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Systems Design Study of the Pioneer Venus Spacecraft

Final Study Report

Volume II. Preliminary Program Development Plan

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29 July 1973

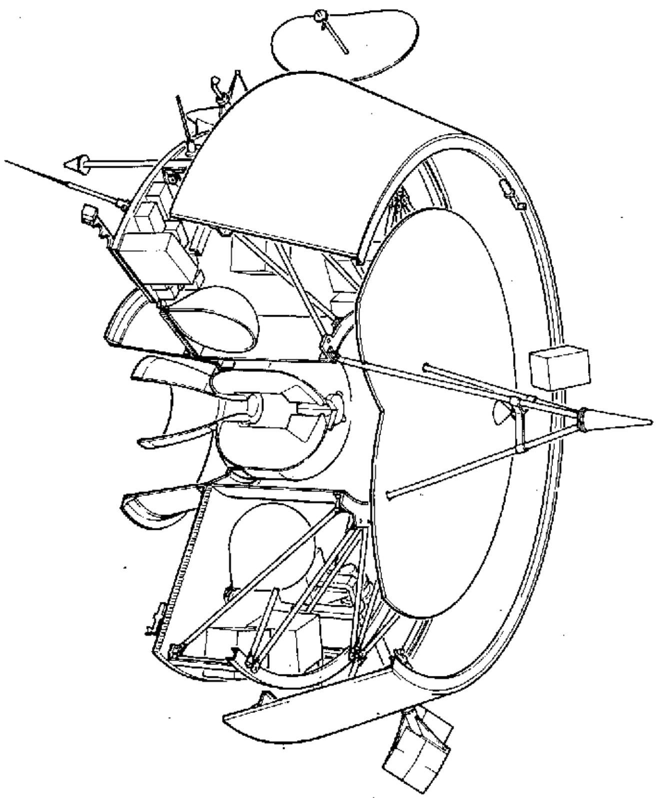
Contract No. NAS2-7249

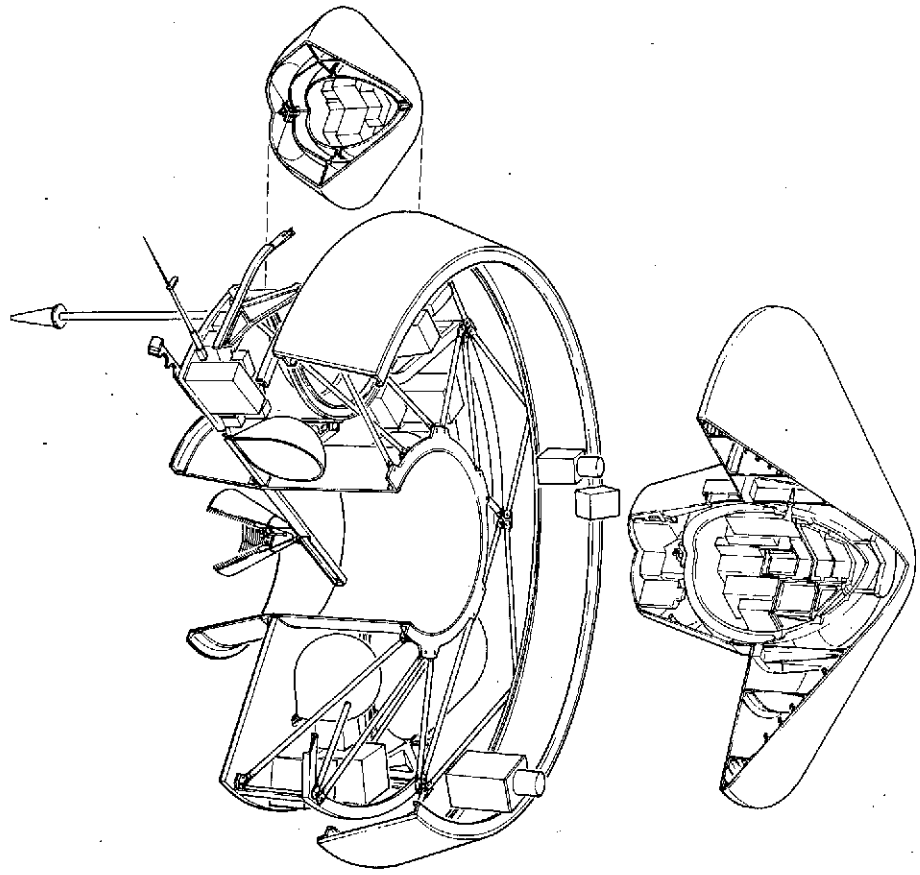
Prepared for

AMES RESEARCH CENTER
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

TRW SYSTEMS GROUP

MARTIN MARIETTA





LIST OF VOLUMES

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TABLES

ACRONYMS AND ABBREVIATIONS

A	ampere analog
abA	abampere
AC	alternating current
A/C	Atlas/Centaur
ADA	avalanche diode amplifier
ADCS	attitude determination and control subsystem
ADPE	automatic data processing equipment
AEHS	advanced entry heating simulator
AEO	aureole/extinction detector
AEDC	Arnold Engineering Development Corporation
AF	audio frequency
AGC	automatic gain control
AgCd	silver-cadmium
AgO	silver oxide
AgZn	silver zinc
ALU	authorized limited usage
AM	amplitude modulation
a.m.	ante meridian
AMP	amplifier
APM	assistant project manager
ARC	Ames Research Center
ARO	after receipt of order
ASK	amplitude shift key
at. wt	atomic weight
ATM	atmosphere
ATRS	attenuated total refractance spectrometer
AU	astronomical unit
AWG	American wire gauge
AWGN	additive white gaussian noise
B	bilevel
B	bus (probe bus)
BED	bus entry degradation

ACRONYMS AND ABBREVIATIONS (CONTINUED)

BER	bit error rate
BLIMP	boundary layer integral matrix procedure
BPIS	bus-probe interface simulator
BPL	bandpass limiter
BPN	boron potassium nitrate
bps	bits per second
BTU	British thermal unit
C	Canberra tracking station — NASA DSN
CADM	configuration administration and data management
C&CO	calibration and checkout
CCU	central control unit
CDU	command distribution unit
CEA	control electronics assembly
CFA	crossed field amplifier
cg	centigram
c.g.	center of gravity
CIA	counting/integration assembly
CKAFS	Cape Kennedy Air Force Station
cm	centimeter
c.m.	center of mass
C/M	current monitor
CMD	command
CMO	configuration management office
C-MOS	complementary metal oxide silicon
CMS	configuration management system
const	constant construction
COSMOS	complementary metal oxide silicon
c.p.	center of pressure
CPSA	cloud particle size analyzer
CPSS	cloud particle size spectrometer

ACRONYMS AND ABBREVIATIONS (CONTINUED)

CPU	central processing unit
CRT	cathode ray tube
CSU	Colorado State University
CTRF	central transformer rectifier filter
D	digital
DACS	data and command subsystem
DCE	despin control electronics
DDA	despin drive assembly
DDE	despin drive electronics
DDU	digital decoder unit
DDULBI	doubly differenced very long baseline interferometry
DEA	despin electronics assembly
DEHP	di-2-ethylhexyl phthalate
DFG	data format generator
DGB	disk gap band
DHC	data handling and command
DIO	direct input/output
DIOC	direct input/output channel
DIP	dual in-line package
DISS REG	dissipative regulator
DLA	declination of the launch azimuth
DLBI	doubly differenced very long baseline interferometry
DMA	despin mechanical assembly
DOF	degree of freedom
DR	design review
DSCS II	Defense System Communications Satellite II
DSIF	Deep Space Instrumentation Facility
DSL	duration and steering logic
DSN	NASA Deep Space Network
DSP	Defense Support Program
DSU	digital storage unit
DTC	design to cost
DTM	decelerator test model

DTP	descent timer/programmer
DTU	digital telemetry unit
DVU	design verification unit
E	encounter entry
EDA	electronically despun antenna
EGSE	electrical ground support equipment
EIRP	effective isotropic radiated power
EMC	electromagnetic compatibility
EMI	electromagnetic interference
EO	engineering order
EOF	end of frame
EOM	end of mission
EP	earth pointer
ESA	elastomeric silicone ablator
ESLE	equivalent station error level
ESRO	European Space Research Organization
ETM	electrical test model
ETR	Eastern Test Range
EXP	experiment
FFT	fast Fourier transform
FIPP	fabrication/inspection process procedure
FMEA	failure mode and effects analysis
FOV	field of view
FP	fixed price frame pulse (telemetry)
FS	federal stock
FSK	frequency shift keying
FTA	fixed time of arrival

G	Goldstone Tracking Station - NASA DSN gravitational acceleration
g	gravity
G&A	general and administrative
GCC	ground control console
GFE	government furnished equipment
GHE	ground handling equipment
GMT	Greenwich mean time
GSE	ground support equipment
GSFC	Goddard Space Flight Center
H	Haystack Tracking Station - NASA DSN
HFFB	Ames Hypersonic Free Flight Ballistic Range
HPBW	half-power beamwidth
htr	heater
HTT	heat transfer tunnel
I	current
IA	inverter assembly
IC	integrated circuit
ICD	interface control document
IEEE	Institute of Electrical and Electronics Engineering
IFC	interface control document
IFJ	in-flight jumper
IMP	interplanetary monitoring platform
I/O	input/output
IOP	input/output processor
IR	infrared
IRAD	independent research and development
IRIS	infrared interferometer spectrometer
IST	integrated system test
I&T	integration and test
I-V	current-voltage

ACRONYMS AND ABBREVIATIONS (CONTINUED)

JPL	Jet Propulsion Laboratory
KSC	Kennedy Space Center
L	launch
LD/AD	launch date/arrival date
LP	large probe
LPM	lines per minute
LPTTL MSI	low power transistor-transistor logic medium scale integration
LRC	Langley Research Center
M	Madrid tracking station - NASA DSN
MAG	magnetometer
max	maximum
MEOP	maximum expected operating pressure
MFSK	M'ary frequency shift keying
MGSE	mechanical ground support equipment
MH	mechanical handling
MIC	microwave integrated circuit
min	minimum
MJS	Mariner Jupiter-Saturn
MMBPS	multimission bipropellant propulsion subsystem
MMC	Martin Marietta Corporation
MN	Mach number
mod	modulation
MOI	moment of inertia
MOS LSI	metal over silicone large scale integration
MP	maximum power
MSFC	Marshall Space Flight Center
MPSK	M'ary phase shift keying
MSI	medium scale integration
MUX	multiplexer
MVM	Mariner Venus-Mars

ACRONYMS AND ABBREVIATIONS (CONTINUED)

NAD	Naval Ammunition Depot, Crane, Indiana
N/A	not available
NiCd	nickel cadmium
NM/IM	neutral mass spectrometer and ion mass spectrometer
NRZ	non-return to zero
NVOP	normal to Venus orbital plane
OEM	other equipment manufacturers
OGO	Orbiting Geophysical Observatory
OIM	orbit insertion motor
P	power
PAM	pulse amplitude modulation
PC	printed circuit
PCM	pulse code modulation
PCM-PSK-PM	pulse code modulation-phase shift keying-phase modulation
PCU	power control unit
PDA	platform drive assembly
PDM	pulse duration modulation
PI	principal investigator proposed instrument
PJU	Pioneer Jupiter-Uranus
PLL	phase-locked loop
PM	phase modulation
p.m.	post meridian
P-MOS	positive channel metal oxide silicon
PMP	parts, materials, processes
PMS	probe mission spacecraft
PMT	photomultiplier tube
PPM	parts per million pulse position modulation
PR	process requirements
PROM	programmable read-only memory
PSE	program storage and execution assembly

ACRONYMS AND ABBREVIATIONS (CONTINUED)

PSIA	pounds per square inch absolute
PSK	phase shift key
PSU	Pioneer Saturn-Uranus
PTE	probe test equipment
QOI	quality operation instructions
QTM	qualification test model
RCS	reaction control subsystem
REF	reference
RF	radio frequency
RHCP	right hand circularly polarized
RHS	reflecting heat shield
RMP-B	Reentry Measurements Program, Phase B
RMS	root mean square
RMU	remote multiplexer unit
ROM	read only memory rough order of magnitude
RSS	root sum square
RT	retargeting
RTU	remote terminal unit
S	separation
SBASI	single bridgewire Apollo standard initiator
SCP	stored command programmer
SCR	silicon controlled rectifier
SCT	spin control thrusters
SEA	shunt electronics assembly
SFOF	Space Flight Operations Facility
SGLS	space ground link subsystem
SHIV	shock induced vorticity
SLR	shock layer radiometer
SLRC	shock layer radiometer calibration

SMAA	semimajor axis
SMIA	semiminor axis
SNR	signal to noise ratio
SP	small probe
SPC	sensor and power control
SPSG	spin sector generator
SR	shunt radiator
SRM	solid rocket motor
SSG	Science Steering Group
SSI	small scale integration
STM	structural test model
STM/TTM	structural test model/thermal test model
STS	system test set
sync	synchronous
TBD	to be determined
TCC	test conductor's console
T/D	Thor/Delta
TDC	telemetry data console
TEMP	temperature
TS	test set
TTL MSI	transistor-transistor logic medium scale integration
TLM	telemetry
TOF	time of flight
TRF	tuned radio frequency
TTM	thermal test model
T/V	thermo vacuum
TWT	travelling wave tube
TWTA	travelling wave tube amplifier
UHF	ultrahigh frequency
UV	ultraviolet

VAC	volts alternating current
VCM	vacuum condensable matter
VCO	voltage controlled oscillator
VDC	volts direct current
VLBI	very long baseline interferometry
VOI	Venus orbit insertion
VOP	Venus orbital plane
VSI	Viking standard initiator
VTA	variable time of arrival
XDS	Xerox Data Systems

1. INTRODUCTION

This volume presents the preliminary development plan for the Pioneer Venus program. Since the effort during the study was directed more at analyzing various developmental approaches rather than defining a given approach, this preliminary plan does not thoroughly treat all developmental aspects, but only those that would have a significant effect on program cost. These significant development areas were:

- Master program schedule planning
- Test planning — both unit and system testing for probes/orbiter/probe bus
- Ground support equipment
- Performance assurance
- Science integration.

During the study, the Martin Marietta Corporation evaluated various test planning options and test method techniques in terms of achieving a low-cost program without degrading mission performance or system reliability. Section 3 of this document defines the approaches studied and the methodology of the selected approach:

- Mission performance requirements — verification technique
- Test requirements — Atlas Centaur/Thor Delta
- Functional test methods
- Development testing
- Manufacturing and acceptance testing.

2. MASTER PROGRAM SCHEDULE PLANNING

2.1 INTRODUCTION

This section discusses the tradeoffs made to optimize the use of available resources as a function of the schedule time available from Phase I go-ahead to launch. Optimization includes the consideration of cost, schedule risk, and compliance with fiscal funding limitations:

- Schedule risk is identified with the time available to recover from a design deficiency and/or a test failure

- Cost factors derive from:

  - Parallelism of tasks — intensive parallel activities that result in higher costs due to decision log jams and the ensuing false starts (decision reversals)

  - Support engineering — the support required to maintain equipment and/or test operations during integration, test, and launch

  - Interval sustaining support — the support required to provide continuity during a lull in design and development — associated with a delayed hardware start (end-loaded) after Phase I.

2.2 PROGRAM PLANS

2.2.1 Program Planning Assumptions

Our planning is based on the following assumptions:

- Phase I would run from 1 February through 30 September 1974, and would culminate in a program system design review that would define all program requirements and interfaces

- Phase II would begin 1 October 1974 and terminate after the launch of the probe mission spacecraft in August 1978

- Project management and technical support must be maintained at an adequate level over the entire 56-month program

- Fiscal funding constraints will result in a low level of available funding during FY 1974 and 1975 periods, and peaking should be avoided.

2.2.2 Program Plans

Three master schedules were developed for analysis. These schedules are shown in Figures 2-1, 2-2, and 2-3 for illustrative purposes; the bars on these schedules do not reflect actual manpower loading.

2.2.2.1 Front-Loaded (Figure 2-1)

This schedule reflects a full startup at the end of Phase I. The startup would include the bus/orbiter and the probes and the approach would be go "get on and get off" as rapidly as possible.

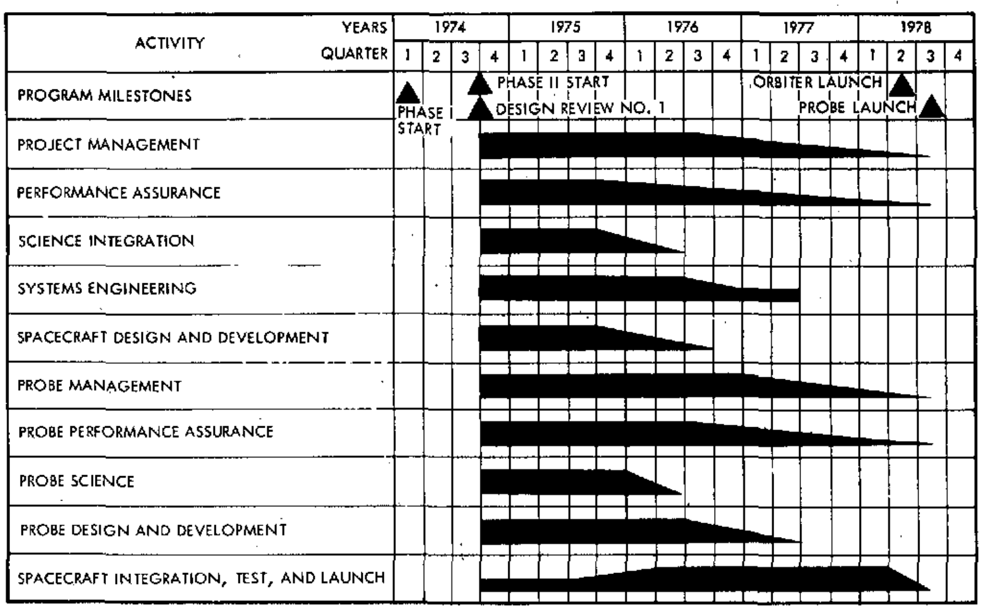

Figure 2-1. Pioneer Venus Project Master Schedule
Case I: Front Loaded

2.2.2.2 Level-Loaded (Figure 2-2)

This schedule is derived from:

- A setback from the launch date to establish hardware need dates
- A "leveling" of the design and development effort in terms of performing: 1) the bus/orbiter design, development and manufacturing effort soon after System Design Review No. 1, and 2) the probe effort later. Thus, the TRW effort builds up moderately fast, while the MMC effort proceeds at a low level in an interface supporting role and in performing major long lead, design critical tasks, and long lead procurement.

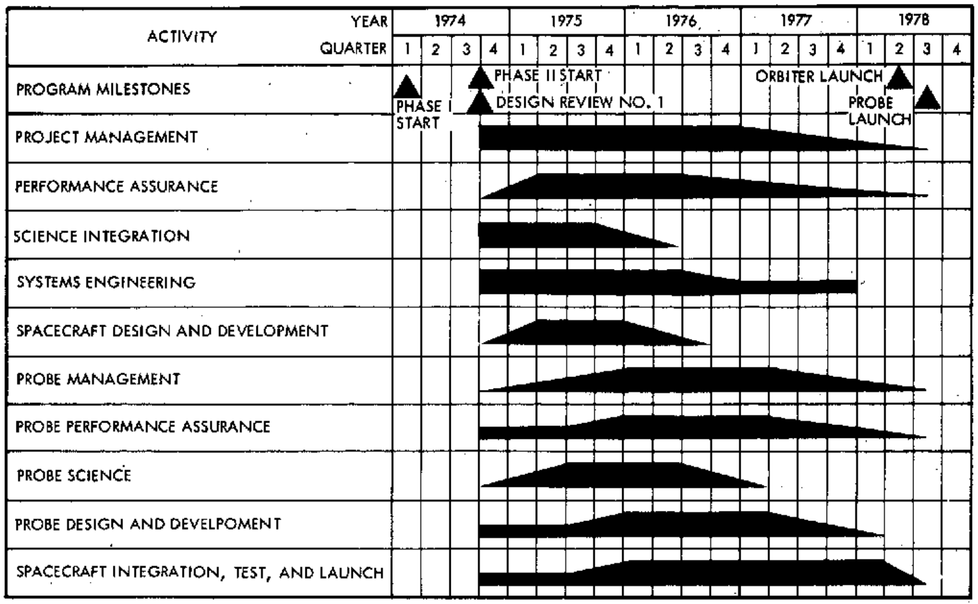

Figure 2-2. Pioneer Venus Project Master Schedule
Case II: Level Loaded

2.2.2.3 End-Loaded (Figure 2-3)

This schedule is based on performing only major long lead and development critical tasks during the period between the completion of System Design Review No. 1 and the startup of the bus/orbiter/probe design and development effort.

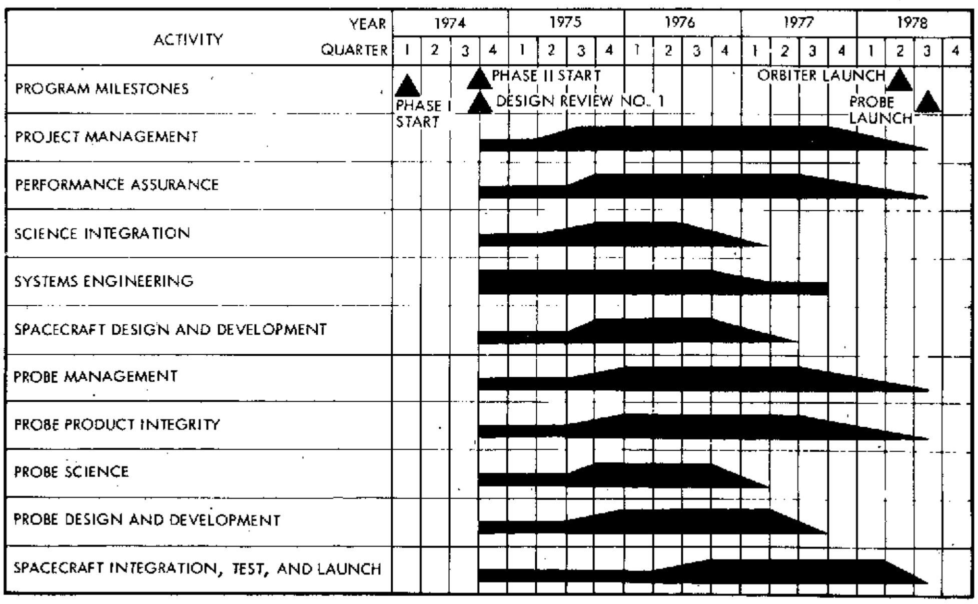

Figure 2-3. Pioneer Venus Project Master Schedule
Case III: End Loaded

2.2.3 Comparison of Plans

Table 2-1 summarizes the evaluation of these plans in terms of cost, schedule risk, and fiscal loading constraints. As the chart shows, the level-loaded program schedule provides the most satisfactory plans in terms of cost, schedule risk, and fiscal loading. A schedule demonstrating this approach is shown in Figure 2-4.

This schedule was generated by first defining the time required to perform the integration and test activities for the orbiter and probe mission spacecraft (Figures 2-5 and 2-6). Then, these span times were arranged so that only one system test set would be required and that the operations at the launch site would begin on 1 April 1978. The next step was to define the equipment need dates of the spacecraft and to insert the engineering model spacecraft tests, structural model tests, and thermal model tests into the schedule. Thus, the need dates for these engineering models and major development models were set. The time to design, develop, manufacture, and test the required equipments was estimated independently and compared to the need dates. Then, adjustments were made to accommodate any conflicts between need dates and delivery dates.

Table 2-1. Evaluation of Master Program Plans

CRITERIA	FRONT-LOADED	LEVEL-LOADED	END-LOADED
RELATIVE COST	HIGHER THAN LEVEL LOADED BECAUSE EXTREME PARALLELISM OF TASKS CAUSES FALSE STARTS LOWER THAN END LOADED BECAUSE SUSTAINING ENGINEER COSTS ARE LESS THAN THOSE REQUIRED IN THE INTERVAL BETWEEN PHASE I AND START OF DESIGN AND DEVELOPMENT	LEAST—MORE EFFICIENT FLOW OF COMMUNICATION, GOOD MORALE, MINIMUM FALSE STARTS, AND LESS OVERALL SUSTAINING EFFORT AND MINIMAL INTERVAL SUPPORT REQUIRED	HIGHER THAN LEVEL-LOADED BECAUSE OF THE ADDITIONAL TEST SET AND TEST CREW, AND BECAUSE OF THE HIGHER LEVEL OF INTERVAL SUPPORT PRIOR TO STARTINGS
RELATIVE SCHEDULE RISK	LOWEST BECAUSE OF THE TIME LEFT TO RECOVER	MORE THAN ADEQUATE—STILL SUFFICIENT TIME TO RECOVER WITHOUT MAJOR IMPACT	HIGHEST SCHEDULE RISK—MUST EXPEND OVERTIME EFFORT TO RECOVER
FISCAL	POOR—HIGH EXPENDITURE RATE IN THE EARLY PART OF THE PROGRAM	SATISFACTORY—MODERATE BUILD UP IN FISCAL 1975 AND LEVEL THROUGH FY 1976 AND EARLY FY 1977, DECREASING IN THE LATTER HALF OF FY 1977 TO A LOW LEVEL IN FY 1978	MAY PEAK TOO HIGH IN FY 1977, 1978 BECAUSE OF START UP COSTS

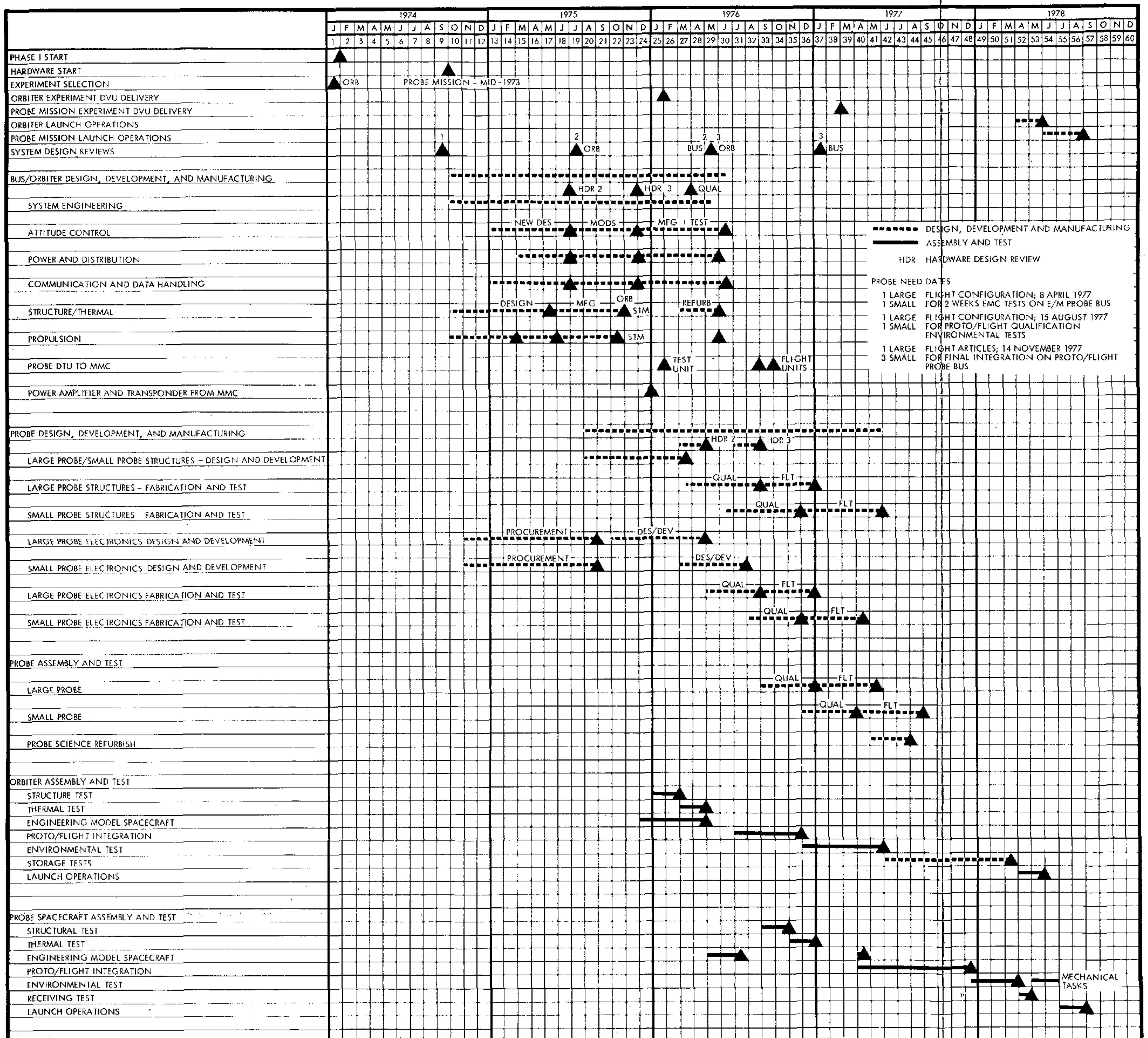

Figure 2-4. Pioneer Venus Schedule

FOLDOUT FRAME /

FOLDOUT FP 2

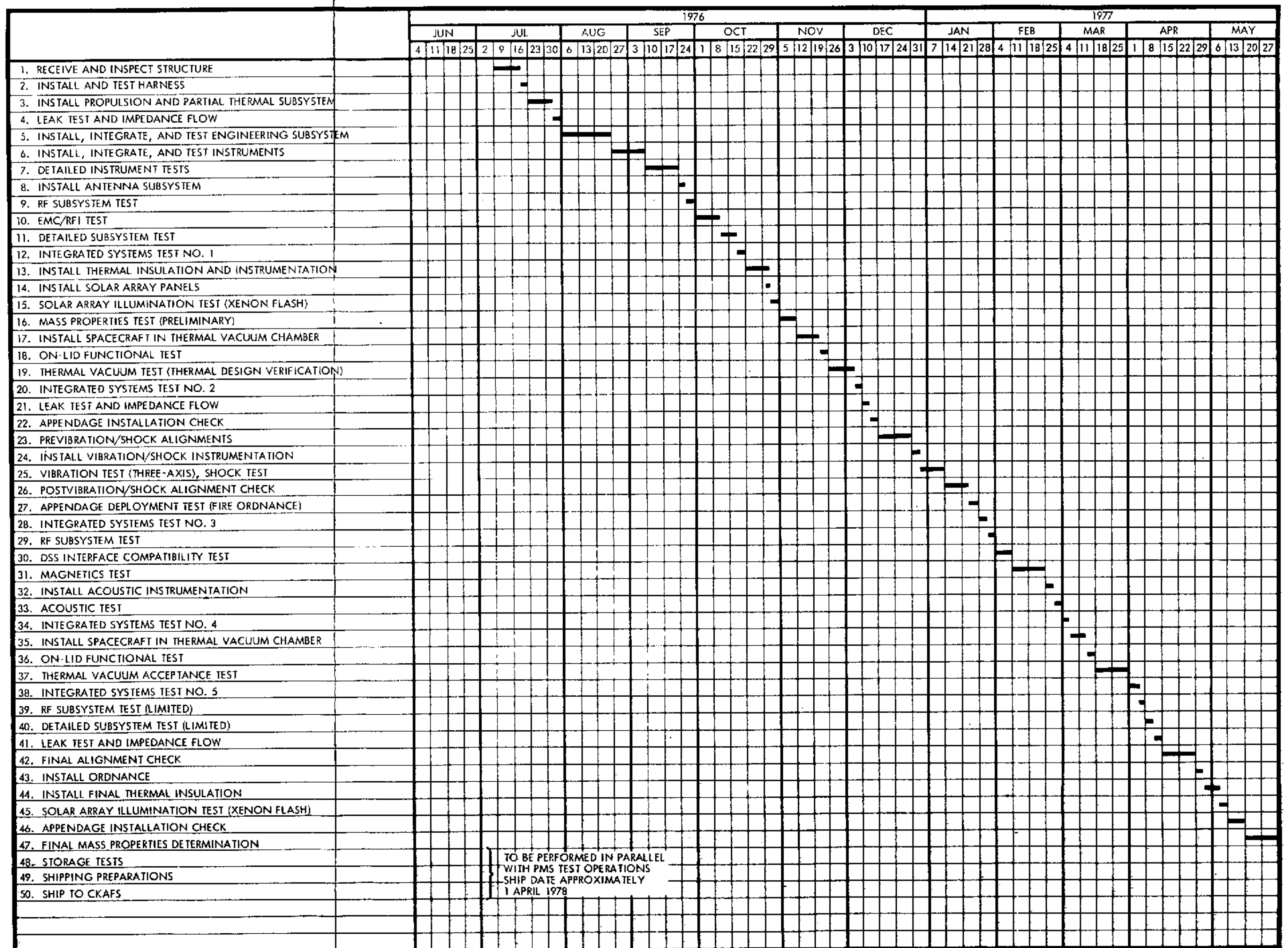

Figure 2-5. Proto/Flight Orbiter Mission Test Schedule

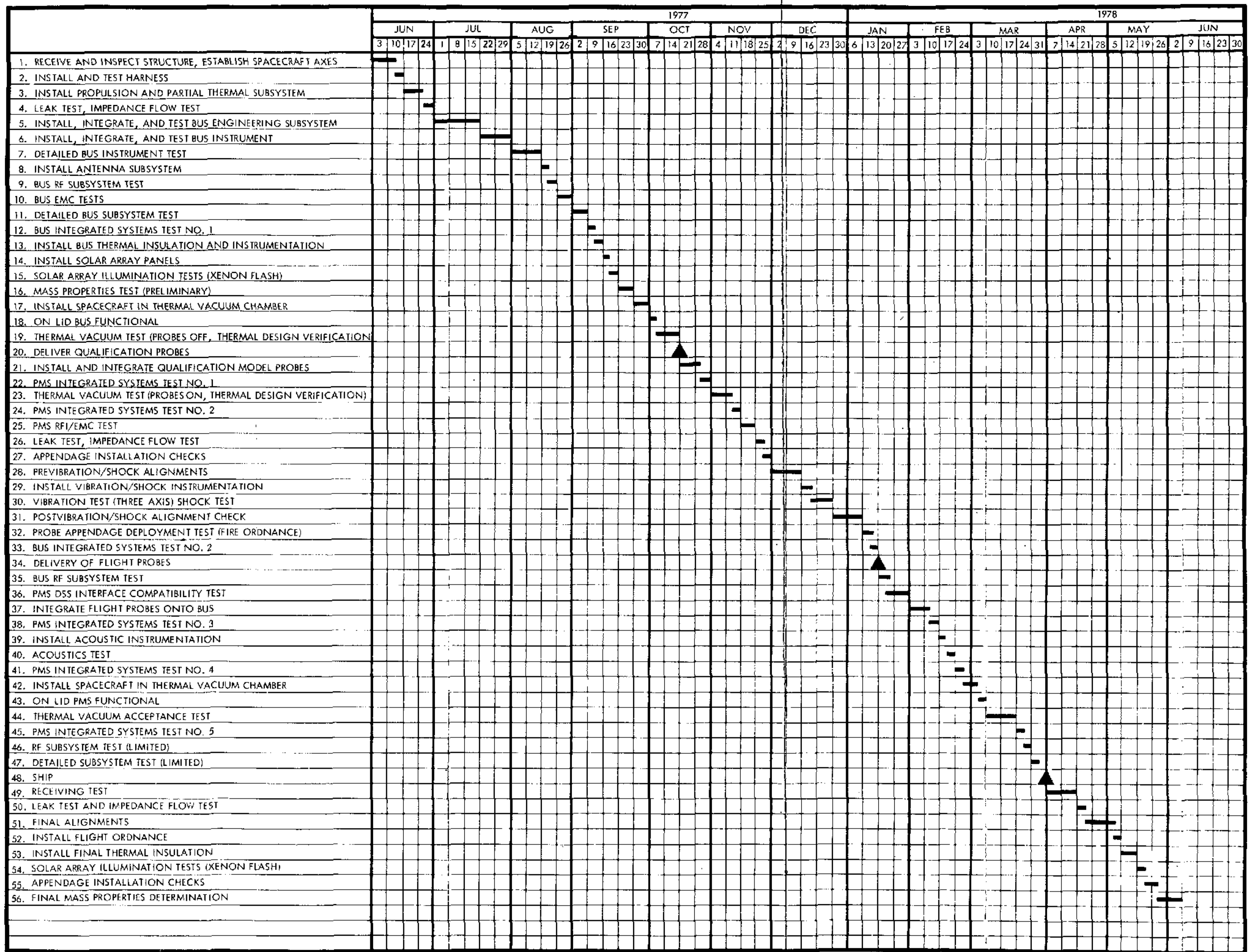

Figure 2-6. Proto/Flight Probe Mission Test Schedule

3. TEST PLANNING

3.1 INTRODUCTION

This section provides a preliminary plan for bus/orbiter subsystems, probe, and system level testing. Relatively little effort was expended in bus/orbiter subsystem/unit level test planning because most of these subsystems/units were derived from the Pioneer 10 and 11 program. Considerable effort was devoted to probe test planning because of the new development required. Several spacecraft system test approaches were studied.

3.2 BUS/ORBITER SUBSYSTEM/UNIT TEST PLANNING

To the extent that Pioneer 10 and 11 hardware/design is used, no test tradeoffs are available between the Thor/Delta and the Atlas/Centaur launch vehicles since that equipment has been qualified to higher levels (Titan 3D). For new design equipment, the weight gained by using the Atlas/Centaur allowed a more conservative design. This resulted in a reduction of the test models required for the Atlas/Centaur.

In planning for the Atlas/Centaur configuration, the following guidelines were applied:

- Maximize the use of Pioneer 10 and 11 test hardware
- Qualify by similarity wherever possible.

The result is shown in the equipment list, Table 3-1, which reflects a minimum cost approach to subsystem level testing without any decrease in confidence.

3.3 PROBE TEST PLANNING

The probes represented a new design in that existing equipment has not been qualified to the extreme environments of Venus atmosphere entry. Therefore, considerable effort was devoted to test tradeoffs for the probe subsystems and probe systems. The objective was to establish probe test requirements and to minimize test hardware to reduce costs. The Atlas/Centaur and Thor/Delta probe test programs established during the study fulfill the requirements to provide design development data, to qualify the large and small probes for the mission environments (including the hostile

Table 3-1. Pioneer Venus Equipment List

* STATUS KEY:
1 - USE AS IS
2 - MINOR MODIFICATION
3 - MAJOR MODIFICATION (RE-QUAL REQUIRED)
4 - NEW DESIGN

MULTIPLE USAGE

SUBSYSTEM: ELECTRICAL POWER AND INTEGRATION

DATE 19 JUNE 1973

PAGE 1 OF 6

NOTES: (M) = MASS OR THERMAL MODEL
(E) = USE PIONEER F/G ENG'G MODEL
(P) = USE PIONEER F/G PROTO. MODEL
(Q) = USE PIONEER F/G QUAL. MODEL
(S) = USE PIONEER F/G SPARE
(NO SUFFIX INDICATES NEW BUILD)

EQUIPMENT	MAKE-BUY	SOURCE	* STATUS	SUPPLIER	SERIAL NUMBER – USAGE: TEST UNIT: EM TEST	TEST UNIT: QUAL TEST	ORBITER: STRUCT MODEL	ORBITER: THERMAL MODEL	ORBITER: ENGR MODEL S/C	ORBITER: PROTO/FLIGHT	PROBE BUS: STRUCT MODEL	PROBE BUS: THERMAL MODEL	PROBE BUS: ENGR MODEL S/C	PROBE BUS: PROTO/FLIGHT	FLIGHT SPARES	QUANTITY: REWORK REFURB. F/G RESID.	QUANTITY: NEW BUILD	QUANTITY: TOTAL	REMARKS
SOLAR PANELS – ORBITER	M	NEW	4	TRW	---	001	(M)	001 THRU 006	---	002 THRU 007	---	---	---	---	001	0	7	7	
SOLAR PANELS – BUS	M	NEW	4	TRW	---	---	---	---	---	---	---	001 THRU 006	---	002 THRU 007	001	0	7	7	QUAL.BY SIM.TO ORBITER
BATTERY – ORBITER	M	NEW	4	TRW	---	---	(M)	(M)	001	002	---	---	---	---	003	0	3	3	
BATTERY – BUS	B	NEW	4	EAGLE PICHER	---	001	---	---	---	---	(M)	(M)	002	003 004	005	0	5	5	S/N003 IS A TEST UNIT
POWER CONTROL UNIT (PCU)	M	P-F/G	3	TRW	001(E)	001(E)	(M)	(M)	001(E)	005(S)	(M)	(M)	001(E)	002(P)	001(Q)	4	0	4	BUS AND ORB. UNITS ARE COMMON EXCEPT FOR ADDITION OF ONE SLICE ON ORB.
SHUNT RADIATOR – ORBITER	M	NEW	4	TRW	---	---	001	001	001	002	---	---	---	---	001	0	2	2	QUAL.ON PROTO/FLIGHT
SHUNT RADIATOR – BUS	M	NEW	4	TRW	---	---	---	---	---	---	001	001	001	002	001	0	2	2	QUAL.ON PROTO/FLIGHT
INVERTER	M	P-F/G	3	TRW	001(Q)	001(Q)	(M)	(M)	001(Q)	008(S)	(M)	(M)	001(Q)	002(P1)	003(P2)	4	0	4	PARTIAL RE-QUAL.
CENTRAL TRF	M	P-F/G	2	TRW	001(E)	---	(M)	(M)	001(E)	004(S)	(M)	(M)	001(E)	002(P)	001(Q)	4	0	4	QUAL.BY SIM.TO PIONEER F/G
SHUNT ELEMENT ASSEMBLY	M	VELA	1	TRW	---	---	(M)	(M)	001 002	002 003	---	---	---	---	001	0	3	3	QUAL.BY SIM. TO VELA
COMMAND DISTRIB. UNIT (CDU)	M	P-F/G	2	TRW	001(E)	---	(M)	(M)	001(E)	004(S)	(M)	(M)	001(E)	002(P)	001(Q)	4	0	4	QUAL.BY SIM.TO P-F/G
HARNESS – ORBITER	M	NEW	4	TRW	---	---	(M)	---	001 (PARTIAL)	002	---	---	---	---	---	0	1+	1+	
HARNESS – BUS	M	NEW	4	TRW	---	---	---	---	---	---	(M)	---	REWORK ORB. S/N001	002	---	0	1	1	

Table 3-1. Pioneer Venus Equipment List (Continued)

* STATUS KEY:
1 - USE AS IS
2 - MINOR MODIFICATION
3 - MAJOR MODIFICATION (RE-QUAL REQUIRED)
4 - NEW DESIGN

MULTIPLE USAGE

SUBSYSTEM: ATTITUDE CONTROL

DATE 19 JUNE 1973

PAGE 2 OF 6

NOTES: (M) = MASS OR THERMAL MODEL
(E) = USE PIONEER F/G ENG'G MODEL
(P) = USE PIONEER F/G PROTO. MODEL
(Q) = USE PIONEER F/G QUAL. MODEL
(S) = USE PIONEER F/G SPARE
(NO SUFFIX INDICATES NEW BUILD)

EQUIPMENT	MAKE-BUY	SOURCE	* STATUS	SUPPLIER	SERIAL NUMBER – USAGE: TEST UNIT		ORBITER				PROBE BUS					QUANTITY			REMARKS
					EM TEST	QUAL TEST	STRUCT MODEL	THERMAL MODEL	ENGR MODEL S/C	PROTO/ FLIGHT	STRUCT MODEL	THERMAL MODEL	ENGR MODEL S/C	PROTO/ FLIGHT	FLIGHT SPARES	REWORK REFURB. F/G RESID.	NEW BUILD	TOTAL	
SUN SENSOR	M	INT.III	3	TRW	001	001	(M)	(M)	001 002	002 003	(M)	(M)	001 004	004 005	001	0	5	5	RE-QUAL.TO NEW TEMP.
CONTROL ELECTRONICS ASSEMBLY (CEA)	M	P-F/G	3	TRW	001(E)	001(E)	(M)	(M)	001(E)	003(S)	(M)	(M)	001(E)	002(P)	001(Q)	4	0	4	
PLATFORM DRIVE	M	FSC	3	TRW	001	001	(M)	(M)	001	003	---	---	---	---	001	0	3	3	DRIVE FOR RAM PLATFORM AND RADAR ALTIMETER

Table 3-1. Pioneer Venus Equipment List (Continued)

* STATUS KEY:

1 - USE AS IS
2 - MINOR MODIFICATION
3 - MAJOR MODIFICATION (RE-QUAL REQUIRED)
4 - NEW DESIGN

MULTIPLE USAGE

SUBSYSTEM: COMMUNICATIONS, DATA AND COMMANDS (PAGE 1 OF 2)

DATE 19 JUNE 1973

NOTES: (M) = MASS OR THERMAL MODEL
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(S) = USE PIONEER F/G SPARE
(NO SUFFIX INDICATES NEW BUILD)

EQUIPMENT	MAKE-BUY	SOURCE	* STATUS	SUPPLIER	SERIAL NUMBER – USAGE											QUANTITY			REMARKS
					TEST UNIT		ORBITER				PROBE BUS								
					EM TEST	QUAL TEST	STRUCT MODEL	THERMAL MODEL	ENGR MODEL S/C	PROTO/ FLIGHT	STRUCT MODEL	THERMAL MODEL	ENGR MODEL S/C	PROTO/ FLIGHT	FLIGHT SPARES	REWORK REFURB. F/G RESID.	NEW BUILD	TOTAL	
CONSCAN PROCESSOR	M	P-F/G	2	TRW	---	---	(M)	(M)	002(P)	005(S)	---	---	---	---	002(P)	2	0	2	QUAL. BY SIM. TO P-F/G
RECEIVER	M	P-F/G	1	TRW	---	---	(M)	(M)	006(S) 007(S)	006(S) 007(S)	(M)	(M)	002(P1) 003(P2)	001(P1) 002(P2)	001(Q)	5	0	5	QUAL. BY SIM. TO P-F/G
TRANSMITTER DRIVER	M	P-F/G	1	TRW	---	---	(M)	(M)	008(S) 003(P2)	008(S) 003(P2)	(M)	(M)	002(P1) 009	002(P1) 009	001(Q)	4	1	5	QUAL. BY SIM. TO P-F/G
POWER AMPLIFIER	B		3	MICRO-WAVE	---	001	(M)	(M)	001 002	002 003	(M)	(M)	001 004	004 005	001	0	5	5	
X-BAND TRANSMITTER	B	MVM 73	1	MOTOR-OLA	---	---	(M)	(M)	001	002	---	---	---	---	001	0	3	3	QUAL. BY SIM. TO MVM
MEDIUM GAIN HORN ANTENNA	M	P-F/G	1	TRW	001(P)	---	001(P)	(M)	001(P)	001(P)	004	(M)	004	004	---	1	1	2	QUAL. BY SIM. TO P-F/G
LARGE OMNI ANTENNA	M	M-35	1	TRW	001	---	001	(M)	---	001	002	(M)	002	002	---	0	2	2	QUAL. BY SIM. TO M-35
SMALL OMNI ANTENNA	M	P-F/G	1	TRW	003(P)	---	003(P)	(M)	003(P)	003(P)	004	(M)	004	004	---	1	1	2	QUAL. BY SIM. TO P-F/G
HIGH GAIN ANTENNA DISH	M	777	1	TRW	001	---	001	(M)	001	001	---	---	---	---	---	0	1	1	QUAL. BY SIM. TO 777
HIGH GAIN ANTENNA FEED	M	P-F/G	1	TRW	001(P)	---	001(P)	(M)	001(P)	001(P)	---	---	---	---	---	1	0	1	QUAL. BY SIM. TO P-F/G
X-BAND HORN ANTENNA	M	777	1	TRW	001	---	001	(M)	001	001	---	---	---	---	---	0	1	1	QUAL. BY SIM. TO 777

Table 3-1. Pioneer Venus Equipment List (Continued)

* STATUS KEY:

1 - USE AS IS
2 - MINOR MODIFICATION
3 - MAJOR MODIFICATION (RE-QUAL REQUIRED)
4 - NEW DESIGN

MULTIPLE USAGE

SUBSYSTEM: COMMUNICATIONS, DATA AND COMMANDS (PAGE 2 OF 2)

DATE 1 JUNE 1973

NOTES: (M) = MASS OR THERMAL MODEL
(E) = USE PIONEER F/G ENG'G MODEL
(P) = USE PIONEER F/G PROTO. MODEL
(Q) = USE PIONEER F/G QUAL. MODEL
(S) = USE PIONEER F/G SPARE
(NO SUFFIX INDICATES NEW BUILD)

EQUIPMENT	MAKE-BUY	SOURCE	* STATUS	SUPPLIER	SERIAL NUMBER – USAGE											QUANTITY			REMARKS
					TEST UNIT		ORBITER				PROBE BUS								
					EM TEST	QUAL TEST	STRUCT MODEL	THERMAL MODEL	ENGR MODEL S/C	PROTO/ FLIGHT	STRUCT MODEL	THERMAL MODEL	ENGR MODEL S/C	PROTO/ FLIGHT	FLIGHT SPARES	REWORK REFURB. F/G RESID.	NEW BUILD	TOTAL	
DIPLEXER	M	P-F/G	1	TRW	---	---	(M)	(M)	001(Q) 008(S1)	008(S1) 009(S2)	(M)	(M)	001(Q) 002(P1)	002(P1) 003(P2)	001(Q)	5	0	5	QUAL. BY SIM. TO P-F/G
RF SWITCH	B	P-F/G	1	TELE-DYNE	---	---	(M)	(M)	004(P2) 007(S1) 008(S2) 010, 011, 012	004(P2) 007(S1) 008(S2) 010, 011, 012	(M)	(M)	013 THRU 017	013 THRU 017	001(Q) 003(P1)	5	8	13	QUAL. BY SIM. TO P-F/G
RF CABLE SET – ORBITER	M	NEW	4	TRW	---	---	(M)	(M)	001	002	---	---	---	---	---	0	2	2	QUAL. ON ENGR MODEL S/C
RF CABLE SET – BUS	M	NEW	4	TRW	---	---	---	---	---	---	(M)	(M)	001	002	---	0	0	2	QUAL. ON ENGR MODEL S/C
DIGITAL TELEMETRY UNIT (DTU)	M	P-F/G	3	TRW	001(Q)	001(Q)	(M)	(M)	001(Q)	005(S)	(M)	(M)	001(Q)	002(P)	001(Q)	3	0	3	
DIGITAL TELEMETRY UNIT – PROBES	M	P-F/G	3	TRW	001	001	---	---	---	---	---	---	---	002 THRU 005	001 006	0	6	6	CFE – TRW TO MMC
DIGITAL DECODER UNIT (DDU)	M	P-F/G	1	TRW	---	---	(M)	(M)	001(Q) 008(S)	008(S) 003(P2)	(M)	(M)	001(Q) 002(P1)	002(P1) 009	001(Q)	4	1	5	QUAL. BY SIM. TO P-F/G
DATA STORAGE UNIT (DSU)	M	NEW	4	TRW	001	001	(M)	(M)	001 THRU 004	002 THRU 005	---	---	---	---	001	0	5	5	

Table 3-1. Pioneer Venus Equipment List (Continued)

* STATUS KEY:

1 - USE AS IS
2 - MINOR MODIFICATION
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4 - NEW DESIGN

[ ] MULTIPLE USAGE

SUBSYSTEM: STRUCTURE/THERMAL

DATE 19 JUNE 1973

NOTES: (M) = MASS OR THERMAL MODEL
(E) = USE PIONEER F/G ENG'G MODEL
(P) = USE PIONEER F/G PROTO. MODEL
(Q) = USE PIONEER F/G QUAL. MODEL
(S) = USE PIONEER F/G SPARE
(NO SUFFIX INDICATES NEW BUILD)

EQUIPMENT	MAKE-BUY	SOURCE	* STATUS	SUPPLIER	SERIAL NUMBER - USAGE											QUANTITY			REMARKS
					TEST UNIT		ORBITER				PROBE BUS								
					EM TEST	QUAL TEST	STRUCT MODEL	THERMAL MODEL	ENGR MODEL S/C	PROTO/ FLIGHT	STRUCT MODEL	THERMAL MODEL	ENGR MODEL S/C	PROTO/ FLIGHT	FLIGHT SPARES	REWORK REFURB. F/G RESID.	NEW BUILD	TOTAL	
BASIC STRUCTURE - ORBITER	M	NEW	4	TRW	---	---	001	001	MOCK-UP	001	---	---	---	---	---	0	1	1	
BASIC STRUCTURE - BUS	M	NEW	4	TRW	---	---	---	---	---	---	001	001	MOCK-UP	001	---	0	1	1	
LARGE PROBE EJECTION MECH.	M	NEW	4	TRW	001	---	---	---	---	---	001	001	---	002	001	0	2	2*	QUAL. ON STRUCT. MODEL
SMALL PROBE EJECTION MECH.	M	NEW	4	TRW	001	---	---	---	---	---	001 002 003	001 002 003	---	002 003 004	001	0	4	4	QUAL. ON STRUCT. MODEL
MAGNETOMETER DEPLOYMENT MECH.	B		3	AIR SUPPLY	001	---	001	001	---	002	---	---	---	---	001	0	2	2	QUAL. ON STRUCT. MODEL
NUTATION DAMPER - ORBITER	M	NEW	4	TRW	---	---	001	001	---	002	---	---	---	---	001	0	2	2	QUAL. ON STRUCT. MODEL
NUTATION DAMPER - HIGH FREQUENCY	M	NEW	4	TRW	---	---	---	---	---	---	001	001	---	002	001	0	2	2	QUAL. ON STRUCT. MODEL
NUTATION DAMPER - LOW FREQUENCY	M	NEW	4	TRW	---	---	---	---	---	---	001	001	---	002	001	0	2	2	QUAL. ON STRUCT. MODEL
INSTRUMENT BOOM	M	NEW	4	TRW	001	---	001	001	---	002	---	---	---	---	001	0	2	2	QUAL. ON STRUCT. MODEL
LOUVER (ASSEMBLIES) ORBITER (6 BLADE)	M	HELIOS	1	TRW	---	001	001	001 THRU 005	---	001 THRU 005	---	---	---	---	---	0	5	5	
LOUVER ASSEMBLIES BUS (4 BLADE)	M	HELIOS	2	TRW	---	---	---	---	---	---	001	001 002	---	001 002	---	0	2	2	
INSULATION - ORBITER (SET)	M	NEW	4	TRW	---	---	---	001	---	002	---	---	---	---	001 (REFURB)	0	2	2	
INSULATION - BUS (SET)	M	NEW	4	TRW	---	---	---	---	---	---	---	001	---	002	001 (REFURB)	0	2	2	

Table 3-1. Pioneer Venus Equipment List (Continued)

* STATUS KEY:

1 - USE AS IS
2 - MINOR MODIFICATION
3 - MAJOR MODIFICATION (RE-QUAL REQUIRED)
4 - NEW DESIGN

☐ MULTIPLE USAGE

SUBSYSTEM: PROPULSION

DATE 19 JUNE 1973

NOTES: (M) = MASS OR THERMAL MODEL
(E) = USE PIONEER F/G ENG'G MODEL
(P) = USE PIONEER F/G PROTO. MODEL
(Q) = USE PIONEER F/G QUAL. MODEL
(S) = USE PIONEER F/G SPARE
(NO SUFFIX INDICATES NEW BUILD)

EQUIPMENT	MAKE-BUY	SOURCE	* STATUS	SUPPLIER	SERIAL NUMBER - USAGE: TEST UNIT: EM TEST	QUAL TEST	ORBITER: STRUCT MODEL	THERMAL MODEL	ENGR MODEL S/C	PROTO/ FLIGHT	PROBE BUS: STRUCT MODEL	THERMAL MODEL	ENGR MODEL S/C	PROTO/ FLIGHT	FLIGHT SPARES	QUANTITY: REWORK REFURB. F/G RESID.	NEW BUILD	TOTAL	REMARKS
ORBIT INSERTION MOTOR	B	INT.III	1	AERO-JET	001	---	001	001	---	002	---	---	---	---	003	0	3	3	S/N001 USED FOR RELIABILITY TEST, THEN INERT-LOADED FOR STRUCT. & THERMAL MODELS QUAL BY SIM. TO INTELSAT
REACTION CONTROL SYSTEM	M	(SEE BELOW)																	QUAL BY SIM. TO P-F/G, 777, FSC
--PROPELLANT TANK	B	777	1	PSI	001	---	(M)	(M)	---	002 003 004	(M)	(M)	---	005 006 007	008	0	8	8	S/N001 USED FOR BURST TEST
--THRUSTER	M	P-F/G FSC	2	TRW	---	---	(M)	(M)	---	001 THRU 008	(M)	(M)	---	009 THRU 016	017	0	17	17	
--PROPELLANT FILTER	B	P-F/G	1	WINTEC	---	---	(M)	(M)	---	001	(M)	(M)	---	002	003(S)	1	2	3	
--PRESS. TRANSDUCER	B	P-F/G	1	STATHAM	---	---	(M)	(M)	---	001	(M)	(M)	---	002	003(S)	1	2	3	
--FILL/DRAIN VALVE	M	P-F/G	1	TRW	---	---	(M)	(M)	---	001	(M)	(M)	---	002	003(S)	1	2	3	
--LINES/FITTINGS (SETS)	M	P-F/G	2	TRW	---	---	(M)	(M)	---	001	(M)	(M)	---	002	---	0	2	2	
--PROPELLANT VALVE	B	P-F/G	2	ALLEN	---	---	(M)	(M)	---	001 THRU 008	(M)	(M)	---	009 THRU 016	017	0	17	17	
--TEMP. TRANSDUCER	B	NEW	4	ROSE-MOUNT	---	---	(M)	(M)	---	001	(M)	(M)	---	002	003	0	3	3	
ADD: STRUCTURE SUBSYSTEM (SEE PAGE 5)																			
SPACECRAFT ADAPTER	M	NEW	4	TRW	---	---	001	---	---	001	002	---	---	002	---	0	2	2	ADAPTS SPACECRAFT TO GFE BOOSTER ADAPTER
SEPARATION SYSTEM	B		1	MDAC	---	---	001	---	---	002	001	---	---	003	001	0	3	3	INCLUDES SEPARATION CLAMPS, RETENTION SPRINGS AND ORDNANCE

Venus entry and descent environments) and to verify the flight worthiness of the flight probes. These test programs meet these requirements with a minimum of tests and test hardware.

The test programs were developed in a logical process commencing with the mission, design, and interface requirements. The requirements were studied to determine which had to be verified by test, then test concepts and schedules were developed and hardware quantities and utilization and the tradeoff studies for unit and probe level testing and GSE were established.

The Thor/Delta and Atlas/Centaur test programs are similar in many respects:

- The Thor/Delta program requires three basic test models: development, qualification, and flight. It provides a flight backup spare and eliminates the need for environmental tests on the flight probes.

- The Atlas/Centaur program fulfills the same objectives with less tests and test hardware by combining development and qualification tests. One structural/qualification test model is required for the large probe and one for the small probel. Thus, only two test models are required for Atlas/Centaur, resulting in major cost savings.

The Thor/Delta test program is based on a 1977 launch and the Version II science payload. The Atlas/Centaur test program has been updated for the 1978 launch and the Version IV science payload.

3.3.1 Performance Requirements versus Verification

Performance requirements may be verified by test or analysis. The requirements as established by system design, mission modes, and system-induced environments were analyzed for mission criticality. Thus, most severe environment levels were determined.

The results of the tradeoff study of the method for verifying the performance requirements are summarized in Table 3-2. The following sections provide additional tradeoff data on those requirements requiring verification by test.

Table 3-2. Verification Methods: Probe Test Versus Analysis

VERIFICATION REQUIREMENTS	MISSION MODE (APPLICABLE OR WORST CASE)	VERIFICATION				RATIONALE/REMARKS
		ATLAS/CENTAUR		THOR/DELTA		
		LARGE PROBE	SMALL PROBE	LARGE PROBE	SMALL PROBE	
VERIFY:						
1. PROBE FUNCTIONAL PERFORMANCE AND INTERFACE COMPATIBILITY WITH SCIENCE EXPERIMENTS, SPACECRAFT AND DSN	ALL	T	T	T	T	REQUIREMENTS TO BE FULFILLED BY UNIT, PROBE, AND SPACECRAFT TEST LEVELS
2. MISSION LIFE	CRUISE	A	A	A	A	PROBES ARE IN DORMANT NONOPERATING STATE IN MILD ENVIRONMENT DURING INTERPLANETARY CRUISE, WHICH DOES NOT EXCEED SHELF LIFE OF MOST PARTS AND MATERIALS
3. TRANSPORTATION AND STORAGE ENVIRONMENTS: HUMIDITY, SALT, FOG, DUST FUNGUS, RAIN, AND TEMPERATURE-ALTITUDE	TRANSPORTATION AND STORAGE	A	A	A	A	TRANSPORTATION AND STORAGE CONDITIONS CAN BE CONTROLLED TO PROTECT THE PROBE HARDWARE FROM EXPOSURE TO THESE ENVIRONMENTS IN EXCESS OF TOLERABLE LIMITS
4. VIBRATION ENCOUNTERED DURING LAUNCH WITH PROBE ATTACHED TO BUS AND VENUS ENTRY – PROBE ONLY	LAUNCH	T	T	T	T	SINE VIBRATION TEST REQUIRED FOR LAUNCH MODE; ENTRY DYNAMICS ARE MAINLY RANDOM AND ACOUSTIC
5. ACOUSTICS ENCOUNTERED DURING LAUNCH WITH PROBE ATTACHED TO BUS AND VENUS ENTRY – PROBE ONLY	ENTRY	T	T	T	A	SEVERE ENTRY ACOUSTICS REQUIRES TESTS ON PROBES HAVING LARGE FLAT SURFACES; THE THOR/DELTA SMALL PROBE CAN BE VERIFIED BY ANALYSIS
6. ACCELERATION ENCOUNTERED DURING LAUNCH AND DECELERATION ENCOUNTERED DURING VENUS ENTRY	ENTRY	T	T	T	T	LAUNCH ACCELERATION IS LESS THAN 5% OF ENTRY DECELERATION PERMITTING THE STRUCTURE TO BE VERIFIED BY ANALYSIS FOR ACCELERATION; CERTAIN UNITS MAY REQUIRE ACCELERATION TEST; DECELERATION TESTS ARE REQUIRED FOR PROBES AND UNITS
7. SOLAR VACUUM ENCOUNTERED DURING CRUISE WITH PROBES ATTACHED TO BUS (EARTH TO VENUS) AND PROBE ONLY (NEAR VENUS)	POSTSEPARATION AND CRUISE	A	T	T	T	THERMAL CONTROL DESIGN MARGINS FOR ATLAS/CENTAUR LARGE PROBE PERMIT VERIFICATION BY SIMILARITY FOR POST-SEPARATION CRUISE; THERMAL VACUUM TEST SPACECRAFT VERIFIES PRESEPARATION CRUISE
8. METEOROID IMPINGEMENT AND PARTICLE RADIATION ENCOUNTERED DURING PRESEPARATION AND POST-SEPARATION CRUISE	CRUISE	A	A	A	A	PREVIOUS PROGRAMS HAVE REVEALED THAT THESE ENVIRONMENTS CAN BE VERIFIED BY ANALYSIS; COMPUTER PROGRAMS EXIST TO SHOW PROBABILITY OF METEOROID IMPINGEMENT; SHIELDING FROM RADIATION CAN BE ANALYZED FOR ADEQUACY
9. HIGH PRESSURES ENCOUNTERED DURING VENUS DESCENT	DESCENT	T	T	T	T	TESTS ARE REQUIRED TO DEMONSTRATE CAPABILITY OF PRESSURE VESSELS TO WITHSTAND HIGH PRESSURE WITH DESIGN MARGIN
10. HIGH TEMPERATURES ENCOUNTERED DURING VENUS DESCENT	DESCENT	A	T	T	T	TESTS ARE REQUIRED TO DEMONSTRATE REQUIREMENT EXCEPT ATLAS/CENTAUR LARGE PROBE, WHICH CAN BE VERIFIED BY ANALYSIS AND A HIGH AND LOW TEMPERATURE TEST WITH PROBE OPERATING AT LEVELS OF TEMPERATURE EXPECTED INTERNAL TO PRESSURE VESSEL
11. PROBE FUNCTIONAL PERFORMANCE DURING TEMPERATURE-PRESSURE PROFILE ENCOUNTER WITH VENUS DESCENT	DESCENT	A	T	T	T	THE FUNCTIONAL PERFORMANCE UNDER COMBINED PRESSURE AND TEMPERATURE DESCENT PROFILE DEMONSTRATES IN SITU PERFORMANCE; THERMAL AND STRUCTURAL DESIGN MARGINS PERMIT THIS REQUIREMENT TO BE VERIFIED BY SIMILARITY FOR ATLAS/CENTAUR LARGE PROBE
12. PYRO SHOCK ENCOUNTERED DURING PROBE DEPLOYMENT, LARGE PROBE STAGING, AND SCIENCE EXPERIMENT ACTIVATION	DESCENT	T	A	T	A	SMALL PROBES DO NOT HAVE PYRO DEVICES BUT ENCOUNTER PYRO SHOCK AT SEPARATION FROM THE BUS; THIS REQUIREMENT CAN BE VERIFIED BY UNIT SHOCK TESTS AND VERIFIED DURING SPACECRAFT SEPARATION TESTS
13. LARGE PROBE DECELERATOR AND STAGING SEQUENCE	DESCENT	T	NA	T	NA	FIELD TESTS ARE REQUIRED TO DEMONSTRATE DECELERATOR PERFORMANCE AND STAGING SEQUENCE FOR LARGE PROBE (NOT APPLICABLE FOR SMALL PROBE)
14. DEPLOYMENT OF PROBE SCIENCE EXPERIMENTS	DESCENT	T	T	T	T	TESTS REQUIRED AT PROBE LEVEL
15. CAPABILITY OF AEROSHELL TO VENT DURING EARTH LAUNCH AND VENUS ENTRY	LAUNCH ENTRY	T	T	T	T	TESTS AT PROBE LEVEL ARE INEXPENSIVE AND DO NOT WARRANT RISK TO VERIFY PERFORMANCE BY ANALYSIS; VENTING FOR VENUS ENTRY NOT APPLICABLE FOR ATLAS/CENTAUR DUE TO DESIGN MARGIN

T - VERIFY BY TEST; A - VERIFY BY ANALYSIS;
NA - NOT APPLICABLE

3.3.2 Test Requirements

The test requirements for the Thor/Delta and Atlas/Centaur programs are summarized in Table 3-3. This table shows the tests required on each test model, the quantities of each, and the multiple usage The arrows in the quantity blocks show the use of the qualification test model (QTM) as flight backup models.

The Atlas/Centaur program uses the QTM structure initially for probe and spacecraft structural static tests. These tests are similar to those required for the Thor/Delta configuration. The increased structural and thermal design margins are large enough to prevent overstress at the test levels, which are above anticipated flight levels, and still permit the reuse of the structure as a flight backup probe. This concept was not acceptable for the Thor/Delta approach because the test requirements are too close to the ultimate design allowable.

Probe level qualification is achieved by verifying the requirement by test and/or analysis. The test models used for verification are shown in Table 3-3. A review of the QTM columns show those test requirements that are verified on the QTM's. If the requirement is already shown as being verified, it will have been accomplished on the DTM or STM. For example, the "parachute release and aft body separation" test requirement is verified during the test on the Thor/Delta STM. Therefore, it is shown as verified under the Thor/Delta QTM column. This test requirement is shown as "not applicable" on the DTM since the STM test verified the requirement. This allows the DTM descent capsule to be recovered in reusable condition. Several tests were eliminated for the Atlas/Centaur concept by combining the STM and QTM requirements. This is shown in the table by the symbol denoting test eliminated. The test requirement is fulfilled by one test on either the STM or QTM.

The rationale for the probe-test-analysis tradeoffs are discussed in subsequent paragraphs.

3.3.3 Critical Environmental Test Trades

This section discusses the rationale for selecting the tests and test models shown in Table 3-3 for verification of the structural, thermal, and mechanical subsystems, and the qualification of these subsystems and

Table 3-3. Probe Test Requirements Summary

TEST NO.	TEST RATIONALE*	OBJECTIVE/REQUIREMENT															
		TEST PROGRAM:	ATLAS/CENTAUR					THOR/DELTA									
		PROBE:	LARGE			SMALL		LARGE					SMALL				
		QUANTITY:	1	1 →	1	1 →	3	1	1	1	1 →	1	1	1	1	1 →	3
		MODEL:	DTM	QTM	FLT	QTM	FLT	ETM	DTM	STM	QTM	FLT	ETM	STM	TTM	QTM	FLT
1	VACUUM LEAK (REFERENCE SECTION 3.3.3.1)	VERIFY THAT CRUISE (OUTWARD) LEAK RATE IS WITHIN TOLERANCE FOR THE PRESSURE VESSEL SEALS AND SCIENCE AND PENETRATION INTERFACES	E	A	A	A	A	E	E	A	A	A	E	A	A	A	A
2	PRESSURE LEAK (REFERENCE SECTION 3.3.3.1)	VERIFY THAT DESCENT (INWARD) LEAK RATE IS WITHIN TOLERANCE FOR THE PRESSURE VESSEL SEALS AND SCIENCE AND PENETRATION INTERFACES	E	A	B	A	B	E	E	A	A	B	E	A	A	A	B
3	PRESSURE (STRUCTURAL INTEGRITY) (REFERENCE SECTION 3.3.3.1)	VERIFY STRUCTURAL INTEGRITY AND DESIGN MARGIN OF PRESSURE VESSEL TO WITHSTAND HIGH PRESSURES ENCOUNTERED DURING DESCENT	E	A	B	A	B	E	E	A	A	A	E	A	A	A	A
4	AEROSHELL VENT AND BASE COVER PRESSURE (REFERENCE SECTION 3.3.3.2)	VERIFY VENTING OF AEROSHELL FOR EARTH LAUNCH AND VENUS DESCENT MODES AND BASE COVER STRUCTURAL INTEGRITY	E	A	B	A	B	E	E	A	B	B	E	A	E	B	B
5	MASS PROPERTIES (REFERENCE SECTION 3.3.3.8)	VERIFY WEIGHT, CENTER OF GRAVITY, AND ALIGNMENT FOR DESCENT AND CRUISE MODES	A	A	A	A	A	E	A	A	A	A	E	A	A	A	A
6	STATIC (STRUCTURE) ENTRY LOADS (REFERENCE SECTION 3.3.3.2)	VERIFY STRUCTURAL INTEGRITY AND DESIGN MARGIN OF THE STRUCTURE TO WITHSTAND VENUS ENTRY G'S	E	A	B	A	B	E	E	A	B	B	E	A	E	B	B
7	STATIC (STRUCTURE) LAUNCH AND SEPARATION LOADS (REFERENCE SECTION 3.3.3.2)	VERIFY STRUCTURAL INTEGRITY AND DESIGN MARGIN OF THE STRUCTURE TO WITHSTAND LAUNCH AND SEPARATION G'S	E	D	B	D	B	E	E	A	B	B	E	A	E	B	B
8	STATIC (STRUCTURE) PARACHUTE DEPLOYMENT LOAD (REFERENCE SECTION 3.3.3.2)	VERIFY STRUCTURAL INTEGRITY AND DESIGN MARGIN OF THE STRUCTURE TO WITHSTAND PARACHUTE DEPLOYMENT	E	A	B	E	E	E	E	A	B	B	E	E	E	E	E
9	DECELERATION (REFERENCE SECTION 3.3.3.3)	VERIFY STRUCTURAL INTEGRITY OF SMALL PROBE AND LARGE PROBE DESCENT CAPSULE (WITH MARGIN ON THOR/DELTA STM) AND INTERNAL EQUIPMENT TO WITHSTAND VENUS ENGRY G'S (WITH MARGIN ON ATLAS/CENTAUR QTM)	E	A	B	A	B	E	E	A	A	B	E	A	E	A	B
10	FORWARD AEROSHELL RELEASE/PYRO SHOCK (REFERENCE SECTIONS 3.3.3.4 AND 3.3.3.6)	VERIFY CLEAN RELEASE OF FORWARD AEROSHELL AND ABILITY OF PROBE AND INTERNAL EQUIPMENT TO WITHSTAND PYRO SHOCK (AND ESTABLISH SHOCK LEVELS STM)	A	A	B	E	E	E	A	A	A	B	E	E	E	E	E
11	PARACHUTE MORTAR PYRO SHOCK (REFERENCE SECTIONS 3.3.3.4 AND 3.3.3.6)	VERIFY THE ABILITY OF PROBE STRUCTURE AND INTERNAL EQUIPMENT TO WITHSTAND MORTAR PYRO SHOCK AND ESTABLISH SHOCK LEVELS	A	A	B	E	E	E	A	A	A	B	E	E	E	E	E
12	PARACHUTE DEPLOYMENT BRIDLE STRIPOUT AND BASE COVER RELEASE (REFERENCE SECTIONS 3.3.3.4 AND 3.3.3.6)	VERIFY CLEAN STRIPOUT OF PARACHUTE BRIDLE (THOR/DELTA) OR PILOT CHUTE (ATLAS/CENTAUR) FROM AFT COVER. VERIFY RELEASE OF BASE COVER FROM AEROSHELL AFTERBODY FOR ATLAS/CENTAUR.	A	A	B	E	E	E	A	A	B	B	E	E	E	E	E
13	PARACHUTE PERFORMANCE AND PROBE STAGING SEQUENCE (REFERENCE SECTION 3.3.3.4)	VERIFY LARGE PROBE DECELERATOR PERFORMANCE AND STAGING SEQUENCE UNDER FREE-FALL CONDITIONS (AIRCRAFT DROP)	A	B	B	E	E	E	A	E	B	B	E	E	E	E	E
14	PARACHUTE RELEASE, AFT BODY SEPARATION, AND PYRO SHOCK (REFERENCE SECTIONS 3.3.3.4 AND 3.3.3.6)	VERIFY CLEAN SEPARATION OF AFT BODY, RELEASE MECHANISM, AND ABILITY OF PROBE AND INTERNAL EQUIPMENT TO WITHSTAND PYRO SHOCK AND ESTABLISH SHOCK LEVELS	E.	A	B	E	E	E	E	A	A	B	E	E	E	E	E
15	ACOUSTICS (REFERENCE SECTION 3.3.3.5)	VERIFY STRUCTURAL INTEGRITY AND DESIGN MARGIN OF LARGE PROBE STRUCTURE TO WITHSTAND LAUNCH AND ENTRY ACOUSTICS	E	A	B	A	B	E	E	A	B	B	E	E	E	D	D

*SEE SECTION REFERENCED IN PARENTHESES UNDER EACH TEST TITLE FOR FURTHER INFORMATION AND RATIONALE OF ENTRIES SHOWN IN MATRIX FIELD

DTM- DECELERATOR TEST MODEL
QTM - QUALIFICATION TEST MODEL
STM - STRUCTURAL TEST MODEL
ETM - ELECTRICAL TEST MODEL
TTM - THERMAL TEST MODEL
FLT - FLIGHT UNIT

A - TEST
B - REQUIREMENT VERIFIED BY TEST ON OTHER MODELS
C - TEST ELIMINATED DUE TO REDUNDANCY
D - TEST REQUIREMENT FULFILLED BY ANALYSIS
E - REQUIREMENT NOT APPLICABLE

Table 3-3. Probe Test Requirements Summary (Continued)

TEST NO.	TEST RATIONALE*	OBJECTIVE/REQUIREMENT															
		TEST PROGRAM:	ATLAS/CENTAUR					THOR/DELTA									
		PROBE:	LARGE			SMALL		LARGE					SMALL				
		QUANTITY:	1	1 →	1	1 →	3	1	1	1	1 →	1	1	1	1	1 →	3
		MODEL:	DTM	QTM	FLT	QTM	FLT	ETM	DTM	STM	QTM	FLT	ETM	STM	TTM	QTM	FLT
16	VIBRATION (REFERENCE SECTION 3.3.3.5)	VERIFY STRUCTURAL INTEGRITY AND DESIGN MARGIN OF PROBES AND INTERNAL EQUIPMENT TO WITHSTAND LAUNCH AND ENTRY VIBRATION	E	A	B	A	B	E	E	A	A	B	E	A	E	A	B
17	PYRO SHOCK - MASS SPECTROMETER (REFERENCE SECTION 3.3.3.6)	VERIFY ABILITY OF PROBE INTERNAL EQUIPMENT TO WITHSTAND PYRO SHOCK FROM MASS SPECTROMETER VALVING	E	A	B	E	E	E	E	A	A	B	E	E	E	E	E
18	WINDOW COVER RELEASE (REFERENCE SECTION 3.3.3.6)	VERIFY WINDOW COVER RELEASE FOR SMALL PROBE NEPHELOMETER AND NET FLUX RADIOMETER	E	E	E	A	B	E	E	E	E	E	E	A	E	A	B
19	TEMPERATURE PROBE RELEASE (REFERENCE SECTION 3.3.3.6)	VERIFY TEMPERATURE PROBE DEPLOYMENT FOR SMALL PROBE TEMPERATURE PROBE	E	E	E	A	B	E	E	E	E	E	E	A	E	A	B
20	SPIN BALANCE (REFERENCE SECTION 3.3.3.8)	VERIFY DYNAMIC BALANCE OF PROBES FOR SPACECRAFT CRUISE, PROBE CRUISE, AND DESCENT MODES	E	A	A	A	A	E	E	A	A	A	E	A	A	A	A
21	RF SUBSYSTEM (REFERENCE SECTION 3.3.4.3)	VERIFY ANTENNA PATTERN (QTM), VSWR, INSERTION LOSS OF COMPONENTS FOR PROBE RF SUBSYSTEM (PART OF TYPE III TESTS FOR FLIGHT PROBES, SEE BELOW)	E	A	A	A	A	A	E	E	A	A	A	E	E	A	A
22	THERMAL/SOLAR VACUUM (REFERENCE SECTION 3.3.3.7)	VERIFY PROBE PERFORMANCE FOR POSTSEPARATION THERMAL/ SOLAR VACUUM CRUISE MODE	E	D	D	A	B	E	E	E	A	B	E	E	A	A	B
23	DESCENT PROFILE SIMULATION (REFERENCE SECTION 3.3.3.7)	VERIFY PROBE PERFORMANCE UNDER A SIMULATED PROFILE OF HIGH PRESSURE AND TEMPERATURE DURING VENUS DESCENT	E	D	D	A	B	E	E	E	A	B	E	E	E	A	B
24	TEMPERATURE - HIGH AND LOW (REFERENCE SECTION 3.3.3.7)	VERIFY OPERATIONAL TEMPERATURE REQUIREMENTS FOR INTERNAL PROBE ELECTRONICS	E	A	B	D	D	E	E	E	D	D	E	E	E	D	D
25	MAGNETIC THERMAL - THOR/DELTA (REFERENCE SECTION 3.3.3.9)	DETERMINE MAGNETIC PROPERTY CHANGES WITH INCREASES IN TEMPERATURE ENCOUNTERED DURING VENUS DESCENT (SMALL PROBE SCIENCE REQUIREMENT)	E	E	E	E	E	E	E	E	E	E	E	E	E	A	A
26	MAGNETIC CLEANLINESS (REFERENCE SECTION 3.3.3.9)	DETERMINE MAGNETIC REMANENT FIELD AND VERIFY COMPLIANCE WITH SPECIFICATIONS	E	E	E	E	E	A	E	E	A	A	A	E	E	A	A
27	FUNCTIONAL TEST TYPE III (REFERENCE SECTION 3.3.4.3)	VERIFY FUNCTIONAL PERFORMANCE REQUIREMENTS FOR PROBES IN OPEN VESSEL STATE	E	A	A	A	A	A	E	E	A	A	A	E	E	A	A
28	FUNCTIONAL TEST TYPE II (REFERENCE SECTION 3.3.4.4)	VERIFY FUNCTIONAL PERFORMANCE REQUIREMENTS FOR PROBES IN THE CLOSED VESSEL STATE	E	A	A	A	A	A	E	E	A	A	A	E	E	A	A
29	SPACECRAFT STRUCTURAL/THERMAL (REFERENCE SECTION 3.3.3.5)	VERIFY STRUCTURAL AND THERMAL PERFORMANCE OF PROBES IN LAUNCH AND CRUISE MODE WHEN INTEGRATED IN STM/TTM SPACE-CRAFT AT TRW	E	B	B	B	B	E	E	A	B	B	E	A	A	B	B
30	SPACECRAFT INTEGRATION (REFERENCE SECTION 3.3.4.5)	VERIFY FUNCTIONAL PERFORMANCE OF PROBES IN LAUNCH AND CRUISE MODE WHEN INTEGRATED IN ENGINEERING MODEL OR PROTOTYPE SPACECRAFT AT TRW	E	A	B	A	B	E	E	E	A	B	E	E	E	A	B
31	SPACECRAFT FLIGHT (REFERENCE SECTION 3.3.4.5)	VERIFY FUNCTIONAL PERFORMANCE AND FLIGHT WORTHINESS OF PROBES IN LAUNCH AND CRUISE MODE WHEN INTEGRATED IN FLIGHT SPACECRAFT AT TRW	E	E	A	E	A	E	E	E	E	A	E	E	E	E	A
32	LAUNCH OPERATION (REFERENCE SECTION 3.3.4.5)	VERIFY FLIGHT READINESS OF PROBES FOR LAUNCH AT KSC	E	E	A	E	A	E	E	E	E	A	E	E	E	E	A

*SEE SECTION REFERENCED IN PARENTHESES UNDER EACH TEST TITLE FOR FURTHER INFORMATION AND RATIONALE OF ENTRIES SHOWN IN MATRIX FIELD

DTM - DECELERATOR TEST MODEL
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the electrical subsystem to withstand the mission environments. Several of the tests are grouped to avoid redundancy where the same rationale is applicable. The test numbers referenced in Table 3-3 are shown in parentheses.

3.3.3.1 Pressure Vessel Leak and Structural Integrity (1, 2, 3)

Tests are required on all probes to verify that the probe pressure vessels are capable of maintaining an adequate vacuum seal. Flight probes must also be tested to verify an adequate seal for vacuum leaks after they have been opened and resealed prior to launch. Pressure structural integrity and pressure leak tests are not required on flight probes. The pressure vessels and seals are qualified for this environment on the Thor/Delta STM probes and Atlas/Centaur QTM probes. The design safety factor for the Thor/Delta probes are not as large as those for Atlas/Centaur. Thus, qualification test for the Thor/Delta probes must be conducted as the final test on the STM at a test level of 93 atmospheres. The Thor/Delta QTM's are tested at 74 atmospheres. The higher design factor of safety for Atlas/Centaur permits the QTM presure vessels to be tested at 93 atmospheres and still be acceptable as a backup flight probe. The structural integrity of the pressure vessels to withstand the high entry g's are discussed in Section 3.3.3.2.

3.3.3.2 Aeroshell Venting and Structural Integrity (4, 6, 7, 8)

The aeroshell vent and static tests are conducted on the Atlas/Centaur QTM's to verify the structural integrity of the aeroshell and auxilliary structures and the launch vent design. The STM's are used for Thor/Delta instead of the QTM because the test levels at ultimate design loads prohibit the reuse of the structure for a flight backup probe. The most critical and severe test is the static test for entry loads. Emphasis is placed on a static test of the aeroshell that properly distributes a force to simulate the loads with a 25-percent margin. To simulate the entry g's no centrifuge facilities can adequately house the aeroshells except for the Thor/Delta small probes (see Section 3.3.3.3 below). The static test for launch and separation loads, test number 7, can be eliminated for Atlas/Centaur probes and verified by analysis because the structure has a higher factor of safety, and a very small amount of structure (local interface area) is critical for this condition.

3.3.3.3 Deceleration (9)

The probe must survive the entry g's to achieve any of their objectives. Electronic packaging can be verified at the unit level, but probe level tests are required to verify cable harness integrity and installation processes. Also, the assembly of structural elements and thermal insulation must be evaluated for creep as well as structural integrity at the probe level.

The method used for acceleration testing at the probe level is with the satellite centrifuge at the MMC, Orlando division. This machine can adequately house the probe descent capsules, cable harnesses, and auxiliary structure, less the forward aeroshell. (The aeroshell is tested statically as discussed previously.) For the Thor/Delta probes, two design verification tests are required for each size probe:

- The STM with mass simulators, but with flight type cable harnesses installed, would be tested at 125 percent of anticipated entry g's
- The qualification units would be tested at 125 percent to demonstrate the margin required.

The Thor/Delta QTM would be subjected to flight level (100 percent) with the probes in an operating mode to verify probe performance under simulated entry g's. The deceleration test on the Atlas/Centaur QTM's is the same as the Thor/Delta QTM's but at 125-percent levels. The increased factor of safety will permit the QTM's to be flightworthy after the test.

3.3.3.4 Large Probe Parachute Deployment and Staging (10, 11, 12, 13, and 14)

Field tests are required on the large probe to demonstrate the parachute deployment and sequence of events associated with the transition from the entry configuration to the descent configuration. Prior to this test, the decelerator subsystem and staging mechanism will have been qualified. A decelerator test model consisting of a flight-type large probe fore and aft body aeroshell, auxiliary structure, separation hardware and decelerator, and a simulated descent capsule will be dropped from an aircraft at an altitude that will simulate the dynamic pressure anticipated at deployment in the Venus atmosphere. A sequencer and pyro power supply

source will be required to initiate the pyrotechnic devices. It is the purpose of this test to verify the decelerator performance and staging under free-fall conditions. Prior to this field demonstration test, a series of tests will be required on the Atlas/Centaur QTM to verify the performance of the pyrotechnic devices deployment mortar and separation mechanisms under simulated load conditions. These series of tests will be acomplished with the flight-type probe electrical units. The Thor/Delta program required tests on both the STM and QTM. The tests on the STM provided pyro shock data for development and qualification requirements. A shock test bed is used on Atlas/Centaur to obtain this data (see Section 3.3.3.6).

3.3.3.5 Acoustic and Vibration Tests (15, 16 and 29)

Differences in the Atlas/Centaur and Thor/Delta launch vehicle dynamics and the larger, heavier Atlas/Centaur probe configurations have resulted in major changes in the dynamic test requirements. The Thor/Delta configuration requires both sine and random vibration tests conducted at unit and large/small probe level for the STM and QTM probes. This is due to the lightweight structure. Acoustic tests are required on the Thor/Delta large probe due to the large, thin afterbody structure.

The only major structural dynamic test required for the Atlas/ Centaur large probe at MMC is an acoustic test on the QTM. Acoustic testing is preferred to random vibration testing for the large probe for two reasons: 1) because the probe surface area is large, acoustic will input a significant energy level to the structure and internal components as compared to random vibration and 2) the size and weight of the large probe are such that a random vibration test would be excessively difficult and expensive. The combined spacecraft-probe sine vibration test at TRW will verify the integrity of the large and small probes in that environment at applicable frequency ranges.

The major dynamic tests required for the Atlas/Centaur small probe at MMC are random vibration and acoustic tests. Because of the size and interface complexities of the small probe, random environments will have a more pronounced effect on the structure and components than acoustic. An exception is the base cover, which is a relatively large area/low mass structure. An element acoustic test of that structural component is required to verify its structural integrity.

3.3.3.6 Pyrotechnic Shock and Mechanical Devices (10, 11, 12, 14, 17, 18, and 19)

In addition to the large probe staging pyrotechnic shock tests, discussed in Section 3.3.3.4, tests are required to demonstrate that science release mechanisms operate satisfactorily when integrated into the probes. Also, the impact of shock environment on the structure and electronic subsystems must be evaluated. The shock level data is necessary to verify design limit levels determine analytically. For Atlas/Centaur, this initial data is obtained from a shock test bed. For Thor/Delta, it is obtained on the STM. Tests conducted on the QTM's qualify the probes for both pyrotechnic shock and mechanism performance. Margin is achieved by conducting three successive pyro firings for the most severe pyrotechnic device determined during the development tests on the Atlas/ Centaur shock test bed or Thor/Delta STM.

3.3.3.7 Temperature and Pressure (22, 23, and 24)

Combined temperature and pressure environments associated with cruise and descent are to be tested at the probe level. Qualification tests for combined environments are planned for the Atlas/Centaur small probe. The qualification of the Atlas/Centaur large probe is achieved by separate pressure test (Nos. 1, 2, and 3) and temperature tests (No. 24) and analysis by similarity to the small probe. The relaxation of weight constraints for the Atlas/Centaur mission relative to Thor/Delta mission allows an increase in the Atlas/Centaur design margins. This, in turn, allows a reduction in the testing required for the structural and thermal control subsystems to ensure a successful mission. Facilities currently available for the Atlas/Centaur small probe cruise and descent tests are capable of providing combined environments that simulate flight conditions quite closely. Thermal data from the small probe test will be correlated with a detailed analytical thermal model. This correlation, together with the large thermal design margins and a detailed large probe analytical model, ensures a low-risk approach. Further, for both missions, flight acceptance thermal testing will be performed on the bus/probe level. These acceptance tests will verify the workmanship of the probe thermal control subsystem. Also, for the Atlas/Centaur, these tests provide a solar/vacuum test for the large probe.

For the Thor/Delta mission, qualification tests are required for the cruise and descent modes for both the large and small probes. A development test is required for the small probe cruise mode. The thermal control subsystem is evaluated on the small probe thermal test model (TTM). The TTM is of the same configuration as the structural test model (STM) except for the addition of thermal simulation of unit heating. This test is only required on the small probe to obtain design data for validating the thermal analytical model. Since the large probe is similar in construction, the large probe thermal analytical model is validated by similarity. Several development tests have been conducted on the thermal control subsystem, making possible this reduction in thermal testing. Qualification of the thermal control system is achieved on the qualification test models (QTM's). Existing solar vacuum facilities provide a solar beam with excellent simulation for the small probe in a two-sun solar vacuum cruise environment. Data from this test validates the small probe design and thermal control subsystem analytical model. Since the large probe thermal analytical model used similar techniques and computations, the large probe is tested and qualified under thermal vacuum conditions. Infrared heaters will similate the solar input for the large probe cruise tests.

The hostile high pressure and temperature environments during descent are simulated in our hyperthermobaric chamber. This unique facility will qualify both the probes in their descent configuration under anticipated pressure and temperature profiles. For Thor/Delta only, structural integrity tests are required in this facility on the large and small STM's to demonstrate structural design margin at elevated temperatures and to qualify the pressure vessel structure. Thermal control tests are not required on the TTM since adequate design data has been obtained from recent research and development tests. Descent profile simulation tests are required on both the large and small QTM probes to qualify the thermal control subsystem design with a functional probe operating under typical descent mode. The high and low temperature test (No. 24) for the Atlas/Centaur large probe is required to verify performance of the electronic subsystem within the temperature limits provided by the thermal control subsystem. This test is not required for the Atlas/Centaur small

probe or the Thor/Delta large and small probes since they are exposed to the more expensive thermal/solar vacuum and descent profile simulation tests.

3.3.3.8 Mass Determination (5 and 20)

Mass properties and spin balance tests are required for each Atlas/Centaur and Thor/Delta probe. Since weight is a major constraint for Thor/Delta probes, more tests and controls at the subprobe level are required to assure the delivered hardware actual weight data are in accordance with allocations. Also, the Thor/Delta launch and mission operations requirements are more restrictive with respect to principal axis and center of gravity requirements. These requirements would result in more engineering and testing effort than is currently planned for the Atlas/Centaur probe system.

3.3.3.9 Magnetic Cleanliness (25 and 26)

The magnetic cleanliness requirements are applicable for the Thor/Delta mission in accordance with the science payload defined for that mission. Both unit and probe level tests are required to determine that the remanent field characteristics are in accordance with design specifications. Several controls are necessary during the fabrication to alleviate the possibility of embedding materials with magnetic properties into the probe hardware. The magnetic thermal test is applicable for the small probe only because it has a magnetometer.

3.3.4 Functional and Interface

3.3.4.1 Functional Test Methods Considered

Several approaches were considered for the probe systems test techniques. The significant methods considered were:

- Self-Test and Go/No-Go Techniques

  These methods were discarded early in the study due to the severe weight limitation for the Thor/Delta probe configuration, and the cost of additional onboard test hardware requirements did not warrant their use on the Atlas/Centaur probes.

* Automated Computer Checkout

  A method was considered where all commands and events would be addressed in a programmed sequence to the probes. This approach required that all functions or measurements had to be satisfied within the tolerance parameter limits before proceeding to the next function. This method was also discarded because of the simplicity and low number of probe mission commands, and the relatively low quantity of probe data measurements and data rate. Also, additional test measurement sensors or parameter limit circuits would be required to be built into the probe for data feedback to the computer, whicn would impose an additional cost and weight penalty.

* Probe Sequenced Command

  This system with recorded data evaluation appeared most promising because it simulates normal mission activation of the probe. The probe 2-hour timer would take over and provide the same activation of subsystems and formatted data readouts as obtained during normal mission flight checkout. A provision for accelerating the 2-hour time into approximately 20 minutes, or until each format of data is exercised three times, appeared advantageous. This approach not only provided preseparation flight checkout but also appears to be the most effective way of performing ground checkouts during the numerous environmental exposure verifications and the bus/probe interface functional tests.

From the number of probe commands and measurements selected for the mission requirements, it was determined that the same measurements could provide adequate evaluation of test results. All essential information required for the mission such as temperatures, voltages, transmitter output, receiver AGC, interior pressure, etc., plus science data are essentially the same measurements required for the test evaluation during the many performance tests that are repeated on the completed probes.

3.3.4.2 Description of Selected Method

After careful assessment of the mission requirements and the probe design, this probe sequenced command functional test approach was selected. These tests adequately verify system performance and demonstrate repeatability of that performance. The probe level electrical functional test requirements are divided into three categories:

Type III - Probe subsystem tests - same as Type II plus standard equipment with special cables and calibration equipment (open probe pressure vessel) for subsystem tests

Type II - Probe tests using bus/probe interface simulator (BPIS) and the RF probe test set to the system test set or MMC data processing station (closed probe pressure vessel)

Type I - Spacecraft/probe tests via the RF link to the DSN or via the RF probe test set to the spacecraft systems test set (STS) data

3.3.4.3 Type III Test (reference 21 and 27)

The Type III functional (see Figure 3-1) tests provide adequate and thorough tests of each probe subsystem in a progressive sequence until all equipment has been activated and exercised in accordance with the performance design requirements. Design parameters and tolerance limits are verified, and all science and engineering measurements are calibrated. Calibrations are obtained on direct read instruments using built-in test points or special test cables for access to probe signals. These measurements are used for 1) establishing accuracy of data obtained at the telemetry data processing stations and 2) as the baseline reference data for all future Type II and Type I test data.

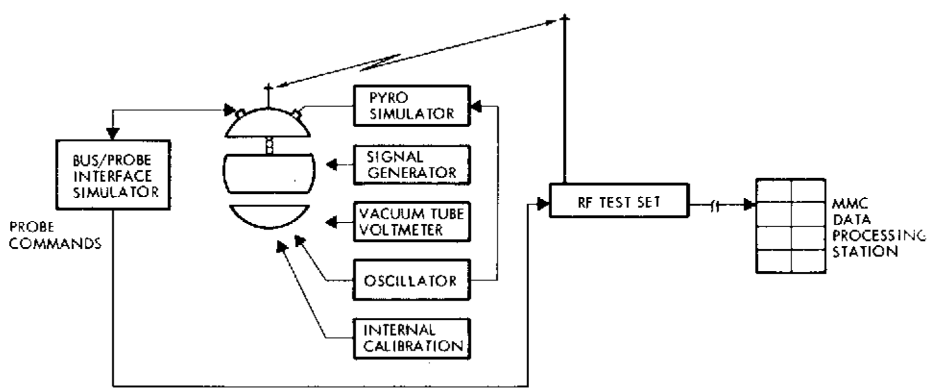


Figure 3-1. Type III - Probe Open Vessel Functional Test

3.3.4.4 Type II (28)

Type II tests (see Figure 3-2) verify the system performance of the probes independent of the spacecraft. The tests are performed after all open pressure vessel Type III tests are completely satisfied. The tests are performed on completely assembled and sealed probes in their pressure vessel or aeroshell configuration. This becomes a standard test that exercises all probe subsystems and science instruments to the maximum

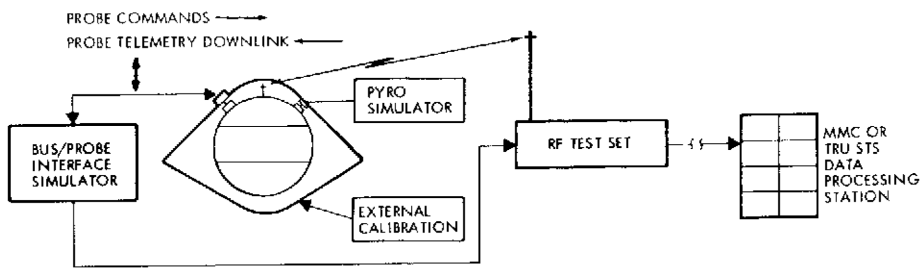


Figure 3-2. Type II - Probe Sealed Vessel (with and without Aeroshell) Functional Test

extent practical with signal access available through the probe connector and RF link. A bus probe interface simulator (BPIS) provides the same command capabilities for addressing and controlling the probe as the bus. All modes, such as real time transmission, mixed transmission (interleaved real-time and stored data), and storage operations, are exercised.

3.3.4.5 Type I (30, 31, and 32)

The Type I tests are the top level probe and spacecraft functional performance tests that verify the design adequacy of the probe. Type I tests (see Figure 3-3) are at the spacecraft level with the probes installed in the bus and tested during a complete spacecraft integrated systems test (IST) operation. The probes are activated in sequence via the spacecraft systems test set (STS). Bus commands and performance data is obtained via the STS or the DSN station.

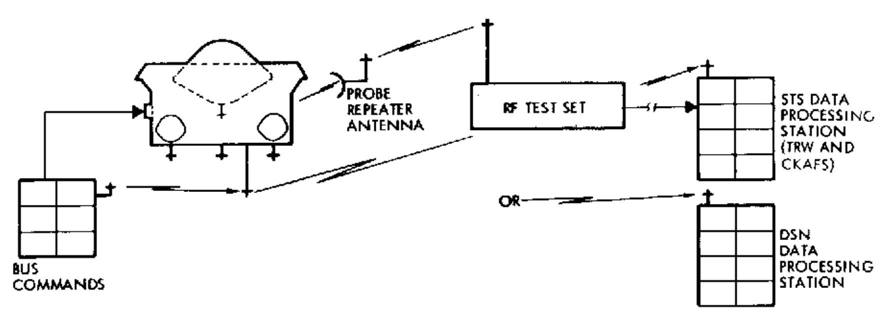


Figure 3-3. Type I - Probes via Spacecraft Functional Test

3.3.5 Major Development Testing

3.3.5.1 Heat Shield

The heat shield is designated as a critical item. Three series of tests on the Atlas/Centaur probe configuration include exploratory tests on 234 material coupon samples, material verification on 30 coupon samples, and verification on 119 coupon samples and realistic sample models of heat shield components and protrusions. The Thor/Delta configuration increases the number of exploratory tests to determine ma terial properties and to verify design adequacy because the heat shield material thickness is constrained by the probe weight limitation.

3.3.5.2 Thermal Control

Component level development tests are required on three thermal control components: phase change devices, joints, and coatings. For the phase change devices, flight-type components will be fabricated to meet area and volume constraints imposed by probe packaging. These devices will be tested to verify their design at the power dissipation levels consistent with the mission. Joint conductance prediction is quite difficult and often predictions are highly inaccurate. Therefore, thermally critical joints for both the large and small probes will be fabricated and tested to provide experimental conductance values. The coatings provide the thermal control during cruise, and the payload temperature is quite sensitive to the solar absorptivity and IR emissivity of the coatings. Tests are required in this area to establish basic surface properties with the Pioneer Venus heat shield material as the substrate and to establish coating degradation data.

3.3.5.3 Wind Tunnel

Atlas/Centaur probe subscale model tests are required to verify and optimize the large and small probe entry vehicle and large probe descent capsule aerodynamic characteristics. Equivalent verification and optimization data is required for the Thor/Delta probe configurations.

3.3.5.4 Electrical Unit Development

Purchased and MMC-developed units require tests to satisfy their adequacy to meet the design requirements. A design verification unit

(DVU) in a final packaged flight configuration (operational prototype) or a qualified unit is required from each supplier to be used in the system integrated tests to evaluate probe electrical compatibility and performance as early as possible. Unit qualification tests are described in Section 3.3.7.

Procurement and test requirement specifications, which are used to initiate the procurement of off-the-shelf buy items and/or buy items to be built to prior proven technology, will be released for procurement after program go-ahead. The items to be purchased include silver-zinc battery cells, diplexer, RF power amplifiers, transponder, decelerator, separation devices, and initiators. Design modifications and the initiation of environment extremes beyond the vendor-specified design limit require verification tests to the new extremes.

Test requirements for developing large and small probe equipment by MMC include the battery, power control unit (PCU), antenna, RF cabling, cabling of engineering and science equipment, and control of separation devices. Breadboard tests are required initially to verify circuit design feasibility; they begin with standard piece parts and special parts are substituted when available.

3.3.6 Qualification Tests

3.3.6.1 Unit Tests

Qualification tests are required on probe units built to flight-released drawings and specifications for the purpose of qualifying the design. The environmental exposures are at levels and durations that exceed the mission requirements to demonstrate a degree of design safety margin. The same functional performance tests conducted on flight units will be conducted on the qualification units. The qualification levels are verified to limits established during development unit tests.

Unit environment test requirements are summarized in Table 3-4 for Thor/Delta and Atlas/Centaur. The commonality of units between the large and small probe for the Atlas/Centaur configuration permits a reduction in five qualification tests. For common units, only one qualification test is required at levels and durations that are the most severe for either the large or small probe.

Table 3-4. Probe Units — Environments Qualified by Test

ATLAS/CENTAUR LARGE AND SMALL PROBE SUBSYSTEMS/UNITS	THOR/DELTA LARGE AND SMALL PROBE SUBSYSTEMS/UNITS	QUALIFICATION TESTS							
		SHOCK	TEMPERATURE HIGH AND LOW	ACCELERATION/ DECELERATION	VIBRATION/SINE AND RANDOM	RELEASE AND DEPLOY	EMC	STRUCTURAL	THERMAL VACUUM
LARGE PROBE	LARGE PROBE								
* BATTERY (2)	BATTERY	T	T	T	T				
* POWER CONTROL UNIT	POWER CONTROL UNIT	T	T	T	T		T		
CUTTER (CABLE)	CUTTER (CABLE)	T	T	T	T	T		T	T
SEPARATION NUT (EXPLOSIVE)	SEPARATION NUT (EXPLOSIVE)	T	T	T	T	T		T	T
	PIN PULLER	T	T	T	T	T		T	T
* DIGITAL TELEMETRY UNIT	DATA HANDLING AND COMMAND	T	T	T	T		T		
* TRANSPONDER	* TRANSMITTER DRIVER	T	T	T	T		T		
* S-BAND POWER AMPLIFIER (2)	S-BAND POWER AMPLIFIER	T	T	T	T		T		
DIPLEXER	DIPLEXER	T	T	T	T				
	RECEIVER								
* ANTENNA	* ANTENNA	T	T	T	T				T
DECELERATOR	DECELERATOR	T	T	T	T	T		T	T
INITIATORS	INITIATORS	T	T	T	T	T		T	T
SMALL PROBE	SMALL PROBE								
	BATTERY	T	T	T	T				
PIN PULLER NONEXPLOSIVE	PIN PULLER NONEXPLOSIVE ITEM	T	T	T	T	T		T	
STABLE OSCILLATOR	STABLE OSCILLATOR	T	T	T	T		T		
	S-BAND POWER AMPLIFIER	T	T	T	T		T		
	DATA HANDLING AND COMMAND	T	T	T	T		T		
	POWER CONTROL UNIT	T	T	T	T		T		

LEGEND: * - COMMON FOR LARGE AND SMALL PROBES. QUALIFICATION TESTS ARE PERFORMED ON ONE UNIT ONLY
T - TEST

3.3.6.2 Atlas/Centaur Probe-Level Tests

One large and one small probe are fabricated using flight probe engineering and manufacturing processes. The probes are subjected to probe-level environmental tests to qualify the design and manufacturing processes for mission environments. Also, functional tests are conducted to validate test procedures and the probe test support equipment for flight acceptance tests. The science experiments are integrated into the the probes and form, fit, and function are verifield. The science experiments are also tested as an integral part of the probe during probe-level environmental tests.

3.3.6.3 Environmental Tests

The Atlas/Centaur large and small probe test flows, Figure 3-4 and 3-5, shows the sequence from the probe assembly to integration tests with the spacecraft at TRW. The series of probe-level environmental tests are conducted in the sequence of environments the flight probes encounter after postseparation cruise, which terminates with descent profile simulation. Existing MMC solar vacuum facilities provide an adequate solar beam with excellent simulation for the small probe in a two-sun solar vacuum cruise environment. Probe-level acceleration tests verify that the design and manufacturing processes produce hardware that will withstand the high deceleration/acceleration environments. These tests are conducted using the large satellite centrifuge facility at the MMC Orlando Division. This centrifuge adequately houses and tests the probes in the descent capsule configuration. The aeroshell and auxillary structure are adequately tested during static tests. The hostile high pressure and

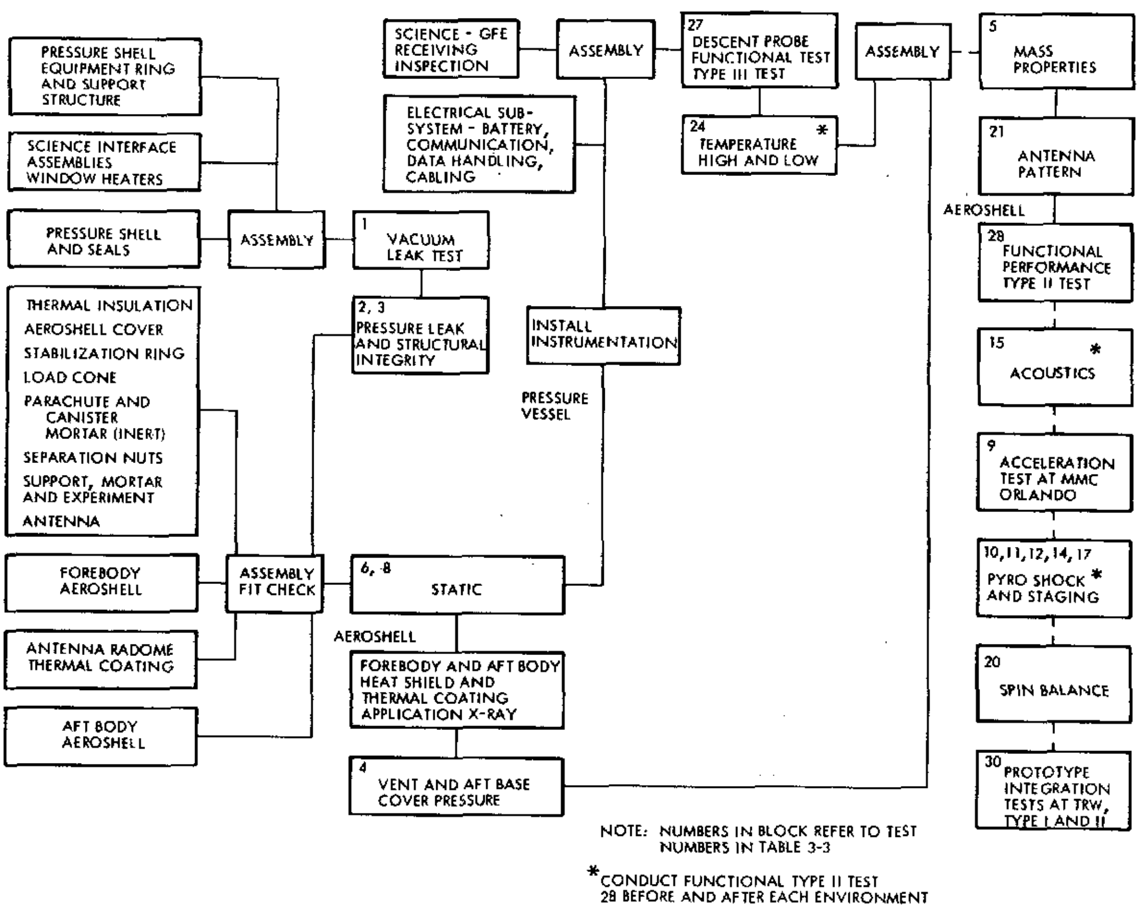
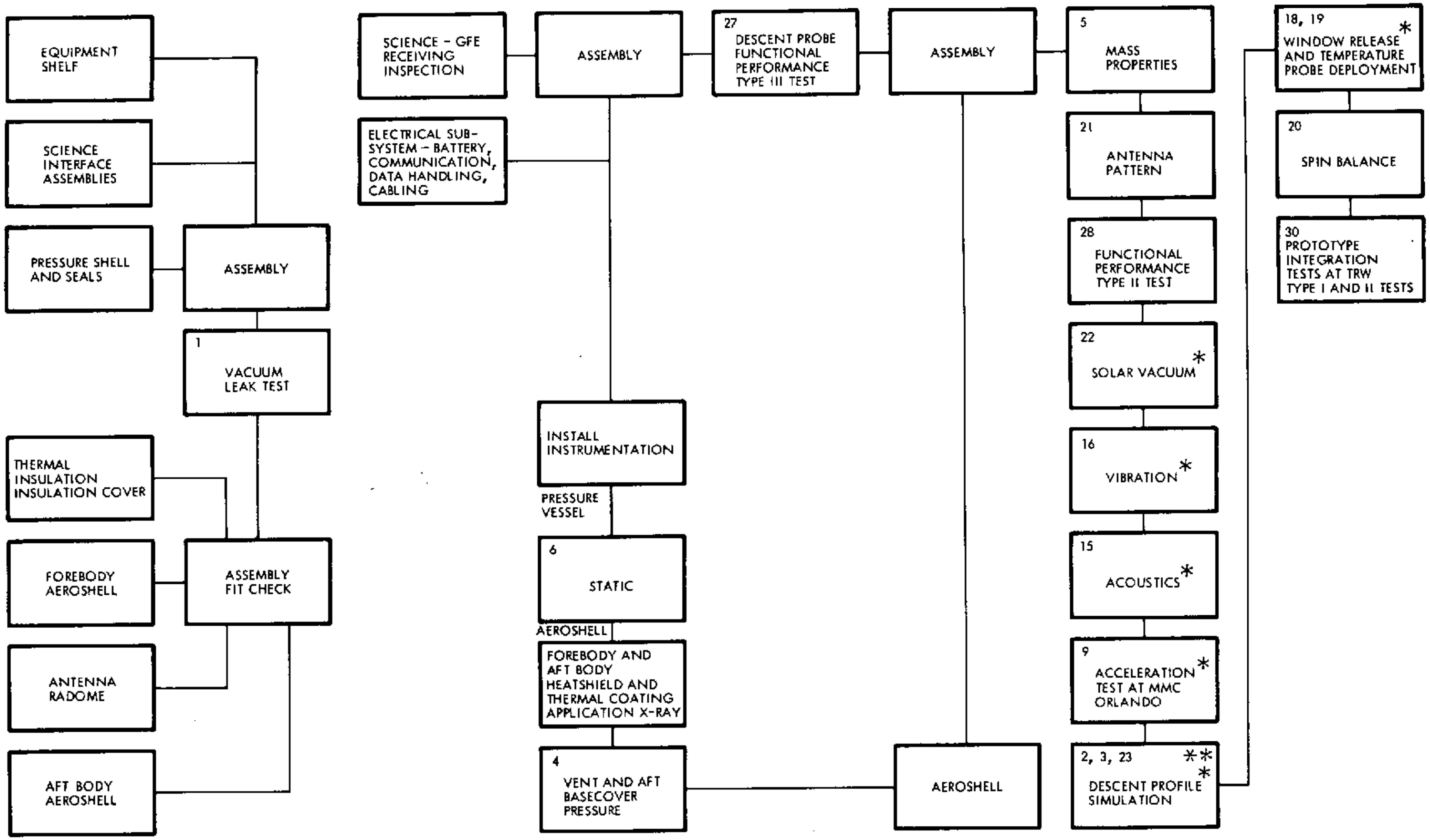


Figure 3-4. Atlas/Centaur Large Probe Qualification Test



NOTE: NUMBERS IN BLOCKS REFER TO TEST NUMBERS IN TABLE 3-3

*CONDUCT FUNCTIONAL TYPE II 28 TEST BEFORE AND AFTER EACH ENVIRONMENTAL TEST

**USE TOOL FOREBODY AND AFT BODY AEROSHELL FOR THIS TEST (HIGH HEAT DEGRADES HEATSHIELD MATERIAL)

Figure 3-5. Atlas/Centaur Small Probe Qualification Test

temperature environments during descent are simulated in the hyperthermobaric chamber, a unique MMC facility. Both large and small probes are tested in their descent configuration at anticipated pressure and temperature profiles.

3.3.7 Acceptance Tests

3.3.7.1 Unit Tests

Functional and environmental tests are required as indicated in Table 3-5. The flight acceptance test includes temperature cycling and vibration tests. These tests are standard MMC production environmental tests for ferreting out manufacturing defects. The temperature cycling requires exposure to at least cycles of high and low temperatures under operating conditions.

The magnetic properties tests are required for Thor/Delta only and are conducted to verify the remanent fields are within tolerance before the next assembly.

Table 3-5. Probe Unit — Acceptance Tests

ATLAS/CENTAUR LARGE AND SMALL PROBE UNITS		THOR/DELTA LARGE AND SMALL PROBE UNITS		FLIGHT ACCEPTANCE TESTS				
				FUNCTIONAL	TEMPERATURE CYCLING	VIBRATION	MASS PROPERTIES	THOR/DELTA MAGNETIC CLEANLINESS
LARGE PROBE		LARGE PROBE						
*	BATTERY	*	BATTERY	T	-	-	T	T
*	POWER CONTROL UNIT		POWER CONTROL UNIT	T	T	T	T	T
	CUTTER (CABLE)		CUTTER (CABLE)	S	-	-	T	T
	SEPARATION NUT (EXPLOSIVE)		SEPARATION NUT (EXPLOSIVE)	S	-	-	T	T
			PIN PULLER	S	-	-	T	T
*	DIGITAL TELEMETRY UNIT	*	DIGITAL TELEMETRY UNIT	T	T	T	T	T
*	TRANSPONDER	*	TRANSPONDER	T	T	T	T	T
*	S-BAND POWER AMPLIFIER (2)	*	S-BAND POWER AMPLIFIER (2)	T	T	T	T	T
	DIPLEXER		DIPLEXER	T	T	T	T	T
*	ANTENNA	*	ANTENNA	T	T	T	T	T
	DECELERATOR		DECELERATOR	T			T	T
	INITIATORS		INITIATORS	S	-	-	T	T
SMALL PROBE		SMALL PROBE						
	PIN PULLER NONEXPLOSIVE ITEM		PIN PULLER NONEXPLOSIVE ITEM	-	-	-	T	T

LEGEND: *COMMON FOR LARGE AND SMALL PROBES
T - TEST
S - TEST REQUIRED ON SAMPLE LOT

3.3.7.2 Atlas/Centaur Probe Acceptance Tests

The acceptance tests for the large and small probes are shown in Figures 3-6 and 3-7. The same tests are also applicable for the flight backup models (refurbished qualification test models). The Type II and III functional tests will adequately verify system performance requirements and demonstrate repeatability of performance, prior to shipping the probes to TRW for spacecraft integration tests. The leak test will verify seal integrity of the pressure vessel during the cruise phase (vacuum). During the mass properties test, which includes weight and center of gravity in the launch, postseparation cruise, and descent configurations, the separation between the Min-K thermal control insulation is packed with F. A. fiberglass insulation. This is the final operation during probe assembly.

3.4 SYSTEMS TEST AND LAUNCH

3.4.1 Introduction

This section describes the integration, test, and launch planning for the probe mission and oribter mission spacecrafts. Alternate plans and schedules are presented and evaluated.

3.4.2 Objectives

The major test objectives for the integration, test and launch phase are to:

- Demonstrate ground support and launch support systems interface compatibility with the spacecraft bus and probe subsystems and scientific instruments
- Demonstrate combined bus/probe and orbiter spacecraft interunit subsystem and scientific instrument interface compatibility
- Demonstrate probe and orbiter mission spacecraft subsystems and system overall performance under all applicable mission conditions through:
  - launch ascent
  - deployment
  - cruise
  - probe release
  - bus/probe encounter - orbiter injection

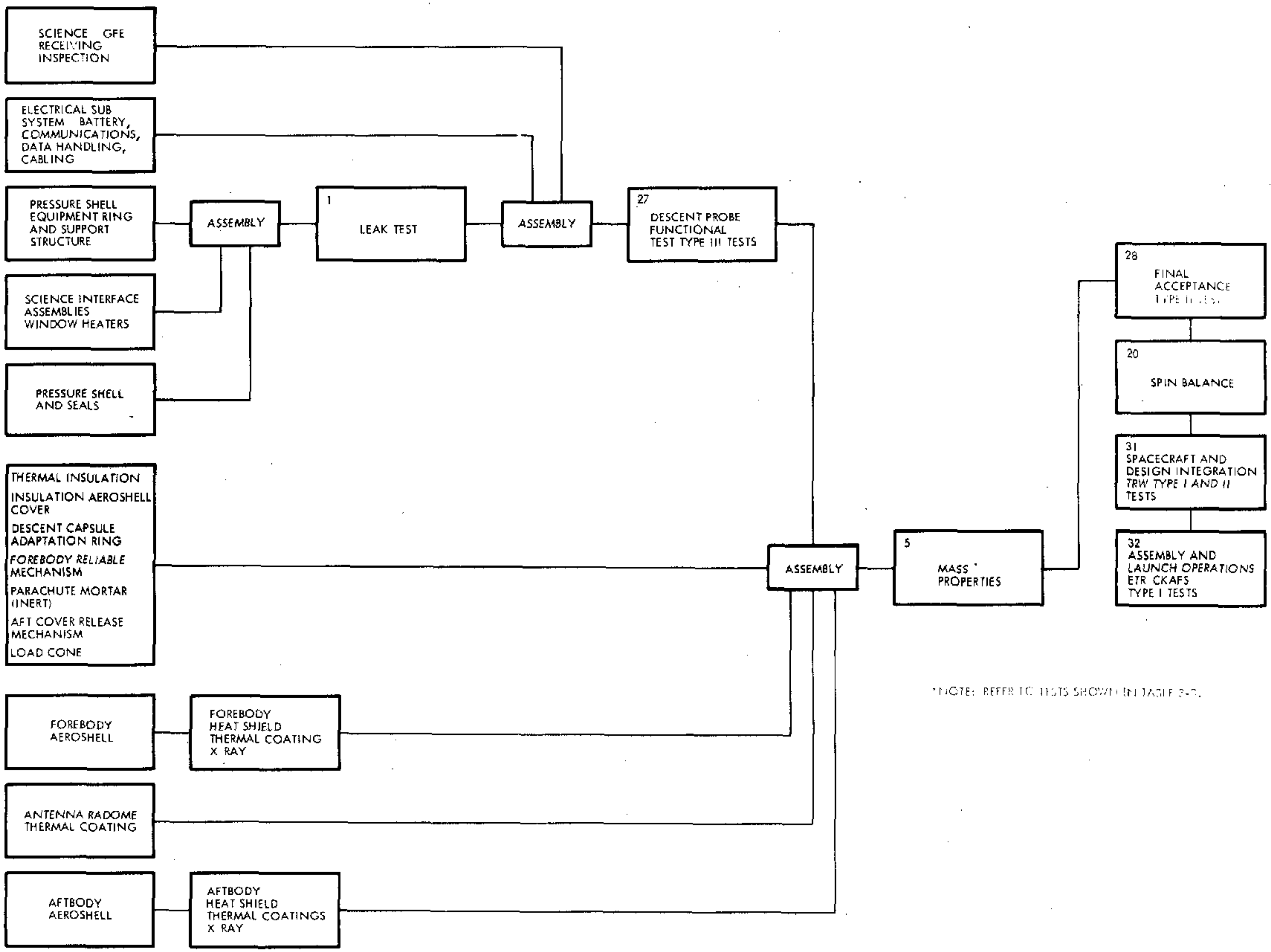


Figure 3-6. Atlas/Centaur Large Flight Probe Test

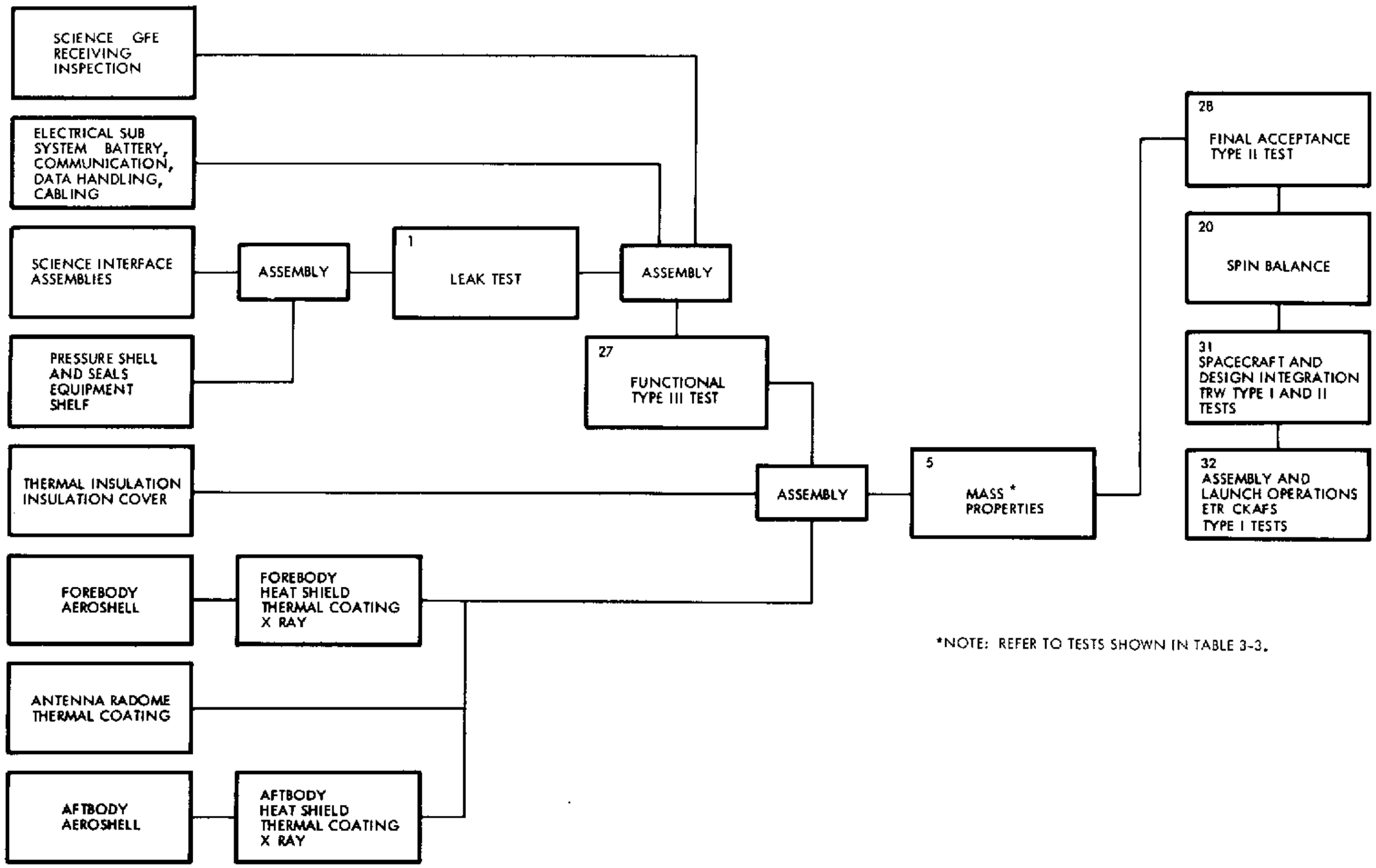


Figure 3-7. Atlas/Centaur Small Flight Probe Test

- Prepare bus/probe and orbiter spacecraft for launch readiness
- Evaluate alternate lower cost plans for achieving these objectives. The evaluation would be in terms of cost versus risk.

3.4.3 Systems Test Tradeoff

3.4.3.1 Evolution of Pioneer Venus Test Plan Philosophy/Requirements

The Pioneers 10 and 11 integration and test plan philosophy was based on the Pioneer 11 spacecraft being virtually a repeat of the 10 spacecraft. This simplified the Pioneer 11 integration and test plan to a repeat flight spacecraft (see Figure 3-8).

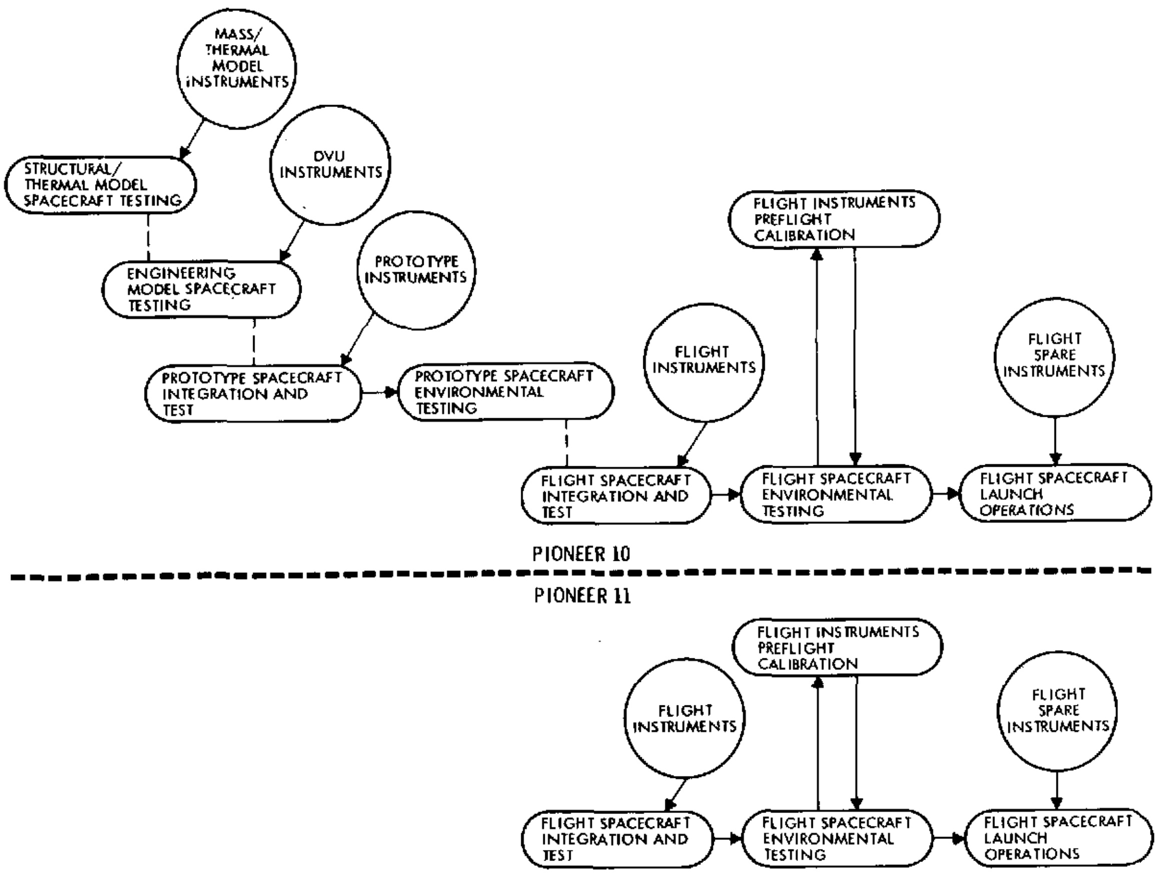


Figure 3-8. Pioneer 10 and 11 Integration Test and Launch Philosophy

The Pioneer Venus probe and orbiter mission spacecraft differ in many respects from one another, resulting in different integration and test requirements compared to Pioneers 10 and 11.

To devise an efficient low-cost integration, test, and launch plan for Pioneer Venus, the following planning objectives were considered:

- Take maximum advantage of TRW's previous test experience with Pioneer hardware and test equipment in meeting all test objecives.
- Reduce integration, test, and launch schedule span times by reducing redundant testing without taking excessive risks
- Plan to minimize risks late in the program
- Plan to reduce hardware costs and minimize "dead-ending" of hardware.

Three integration, test, and launch plans (A, B, and C) were developed. Each of these plans is discussed below, together with accompanying flow charts. Detailed test activities are provided for the preferred plan C. Plans A and B would be similar.

3.4.3.2 Discussion on Alternative Integration, Test, and Launch Plans

Plan A

This plan involves the same test flow as Pioneers 10 and 11 in that it includes the following spacecraft models for the orbiter and probe missions (see Figure 3-9). A major milestone development schedule is shown in Figure 3-10.*

- Probe mission
  - Structural/thermal
  - Engineering model
  - Prototype
  - Flight
- Orbiter mission
  - Structural/thermal
  - Prototype
  - Flight.

*This schedule reflects the use of only one system test set. The probe mission spacecraft is started first, then the orbiter mission spacecraft. This sequence could be reversed.

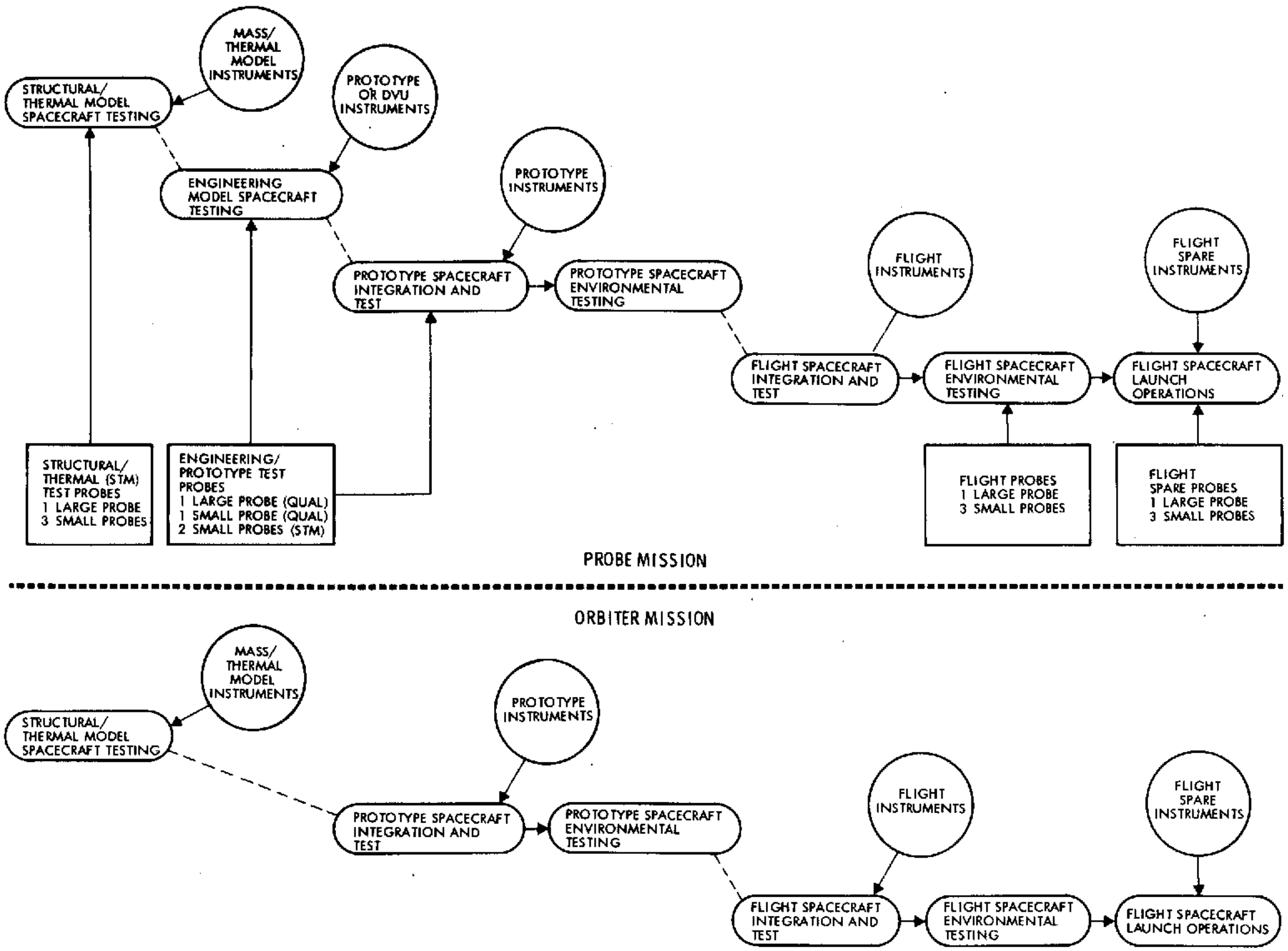


Figure 3-9. Plan A - Pioneer Venus Integration Test and Launch Flow Diagrams

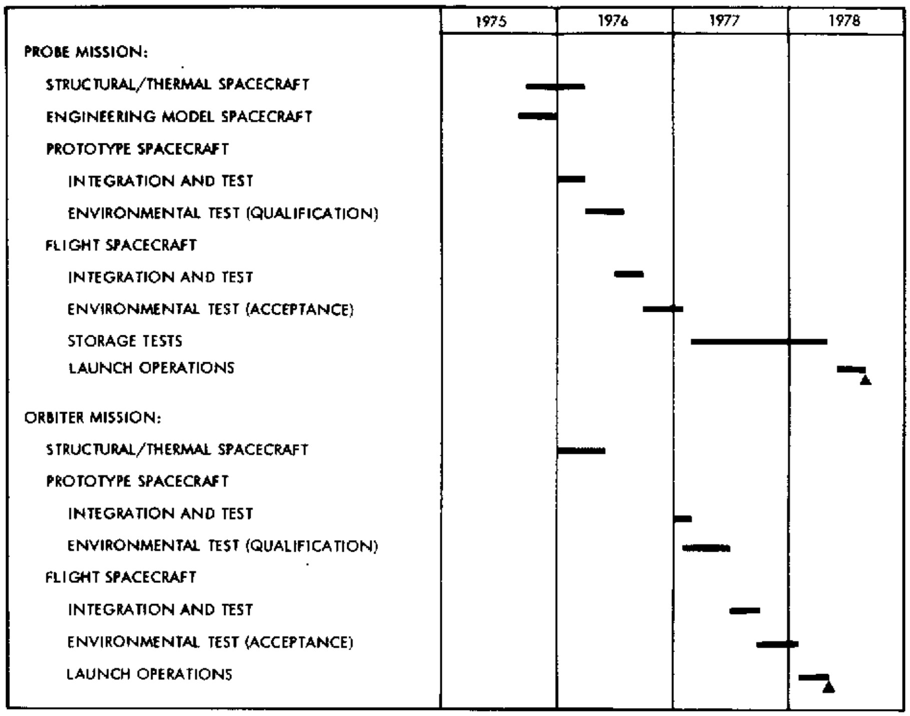


Figure 3-10. Plan A - Probe and Orbiter Mission Integration and Test Schedule

Its key features are:

- Longest overall test schedule and highest cost plan to support
- Does not take maximum advantage of TRW's previous experience in the use of Pioneer 10 and 11 hardware GSE and procedures (repeats spacecraft engineering model testing on many familiar units)
- Lowest average program risk for recovery in case of technical problems, particularly early in the schedule
- Single system test set.

Plan B

The scheduling for plan B allows the engineering model spacecraft test objectives to be satisfied by the prototype spacecraft. The plan includes the following spacecraft models, and a test flow diagram and major milestone schedule as shown in Figures 3-11 and 3-12.

- Probe mission
  - Structural/thermal
  - Prototype
  - Flight

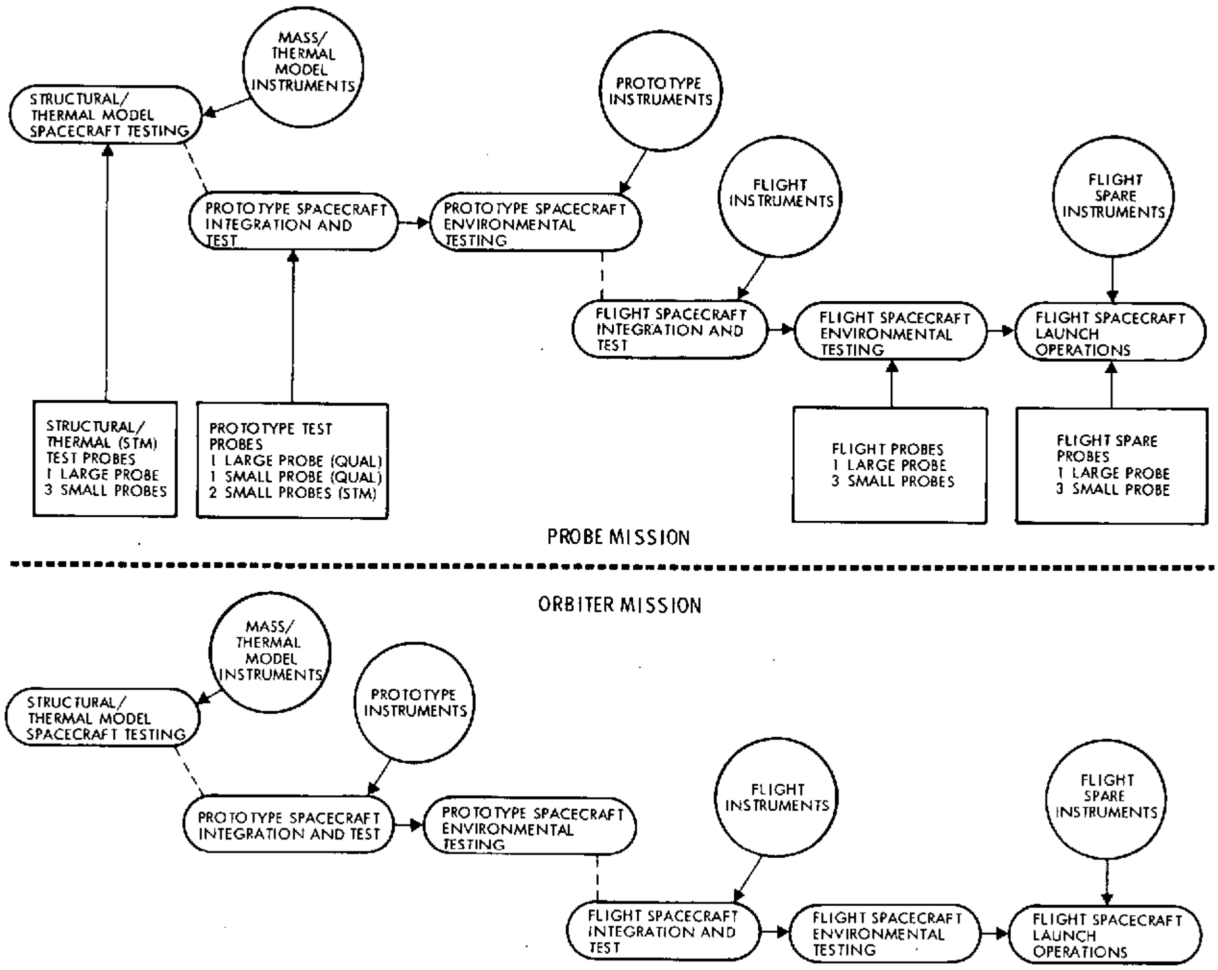


Figure 3-11. Plan B - Pioneer Venus Integration Test and Launch Flow Diagrams

- Orbiter mission
  - Structural/thermal
  - Prototype
  - Flight.

Its key features are:

- Higher schedule risk in case of technical problems
- Reduced integration and test schedule span time for probe mission, resulting in lower program costs than plan A.
- Takes advantage of TRW's previous test experience in use of Pioneer hardware, test equipment, and procedures (engineering and prototype spacecraft testing combined)
- Single system test set.

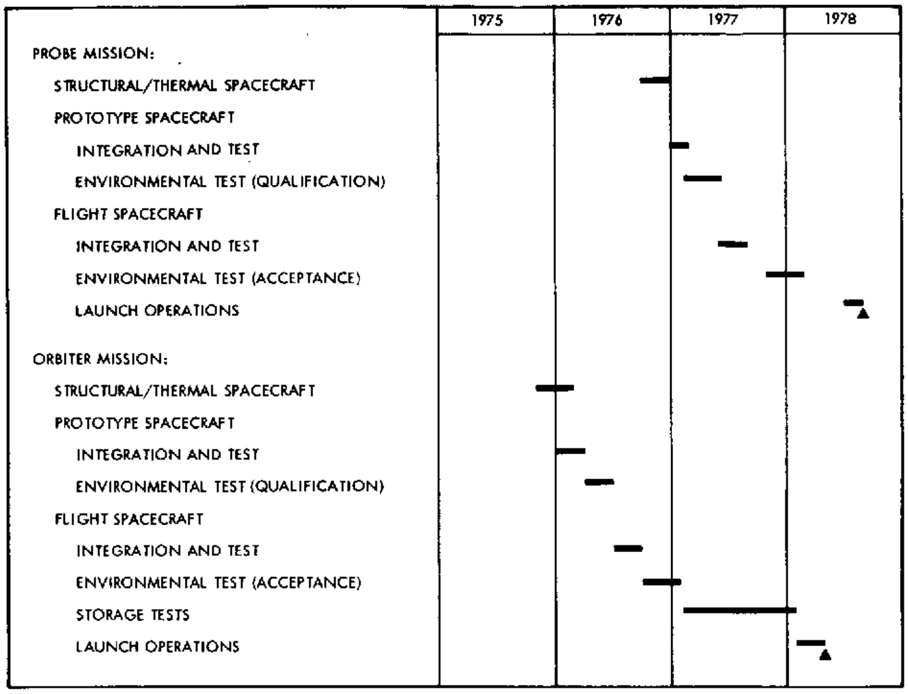


Figure 3-12. Plan B - Probe and Orbiter Mission Integration and Test Schedule

Plan C

The schedule for plan C allows the flight spacecraft (named protoflight) to meet the test objectives of the prototype spacecraft. The plan includes the following spacecraft models for the orbiter and probe mission (see Figure 3-13) and a major milestone development schedule is shown as Figure 3-14.

- Probe mission
  - Structural/thermal
  - Engineering model
  - Prototype/flight
- Orbiter mission
  - Structural/thermal
  - Engineering model
  - Prototype /flight.

Its key features are:

- Potential for performing less environmental test activities by using the protoflight concept; the overall schedule span time remains the same as plan B

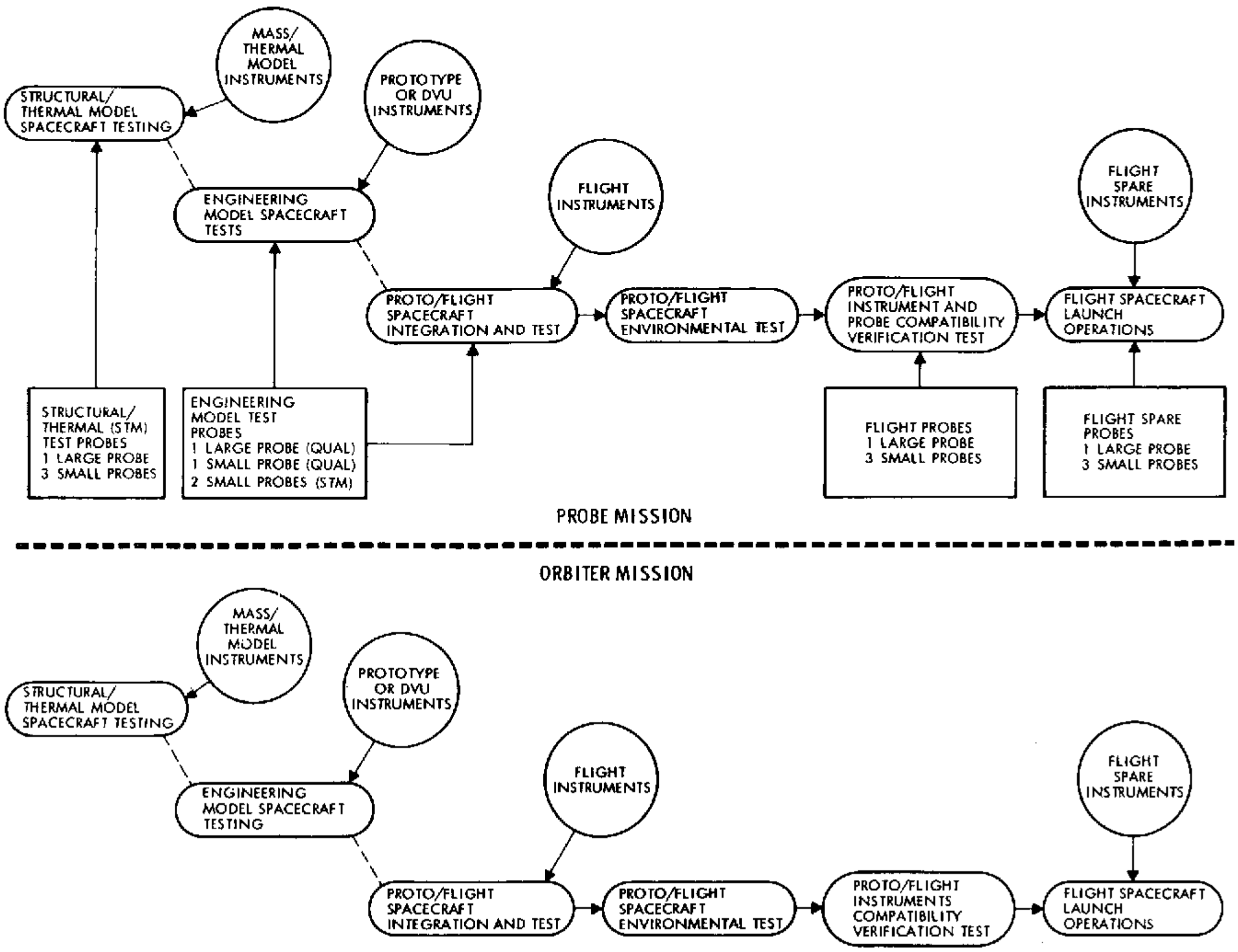


Figure 3-13. Plan C - Pioneer Venus Integration Test and Launch Flow Diagrams

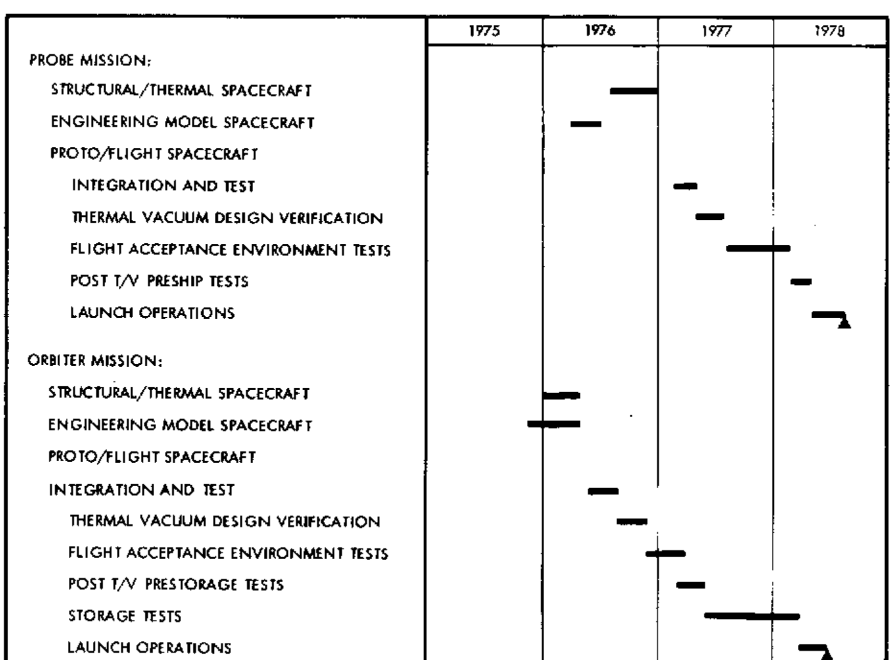


Figure 3-14. Plan C - Probe and Orbiter Mission Integration and Test Schedule

- Schedule provides for potentially less risk than plans A or B late in the schedule
- Takes maximum advantage of TRW's previous test experience in the use of Pioneer hardware, test equipment, and procedures.

Figures 3-15 and 3-16 show the sequence of events for each model. See Figures 2-5 and 2-6 for proto/flight schedules.

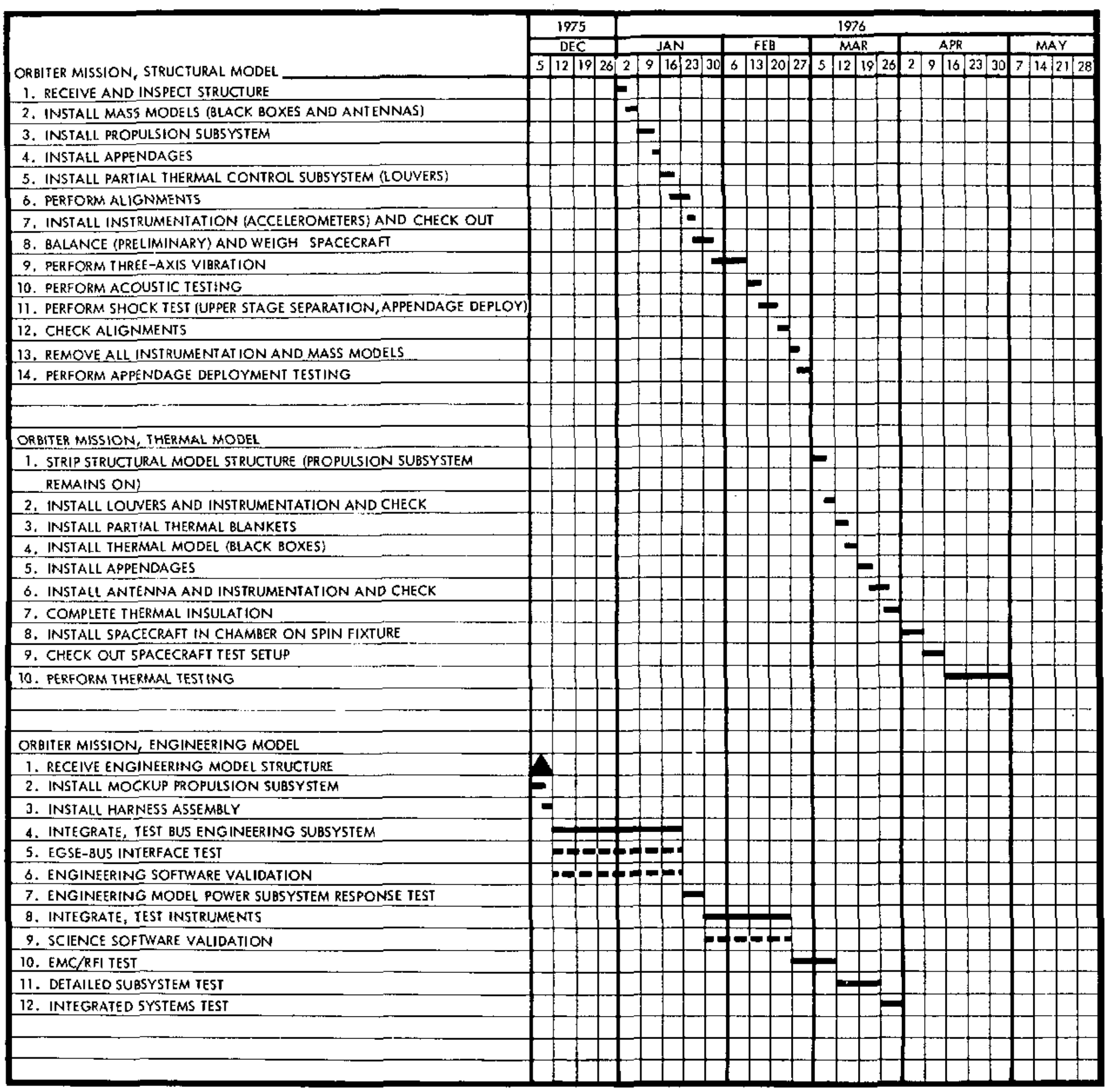


Figure 3-15. Orbiter Mission, Structural, Thermal, and Engineering Model Schedule

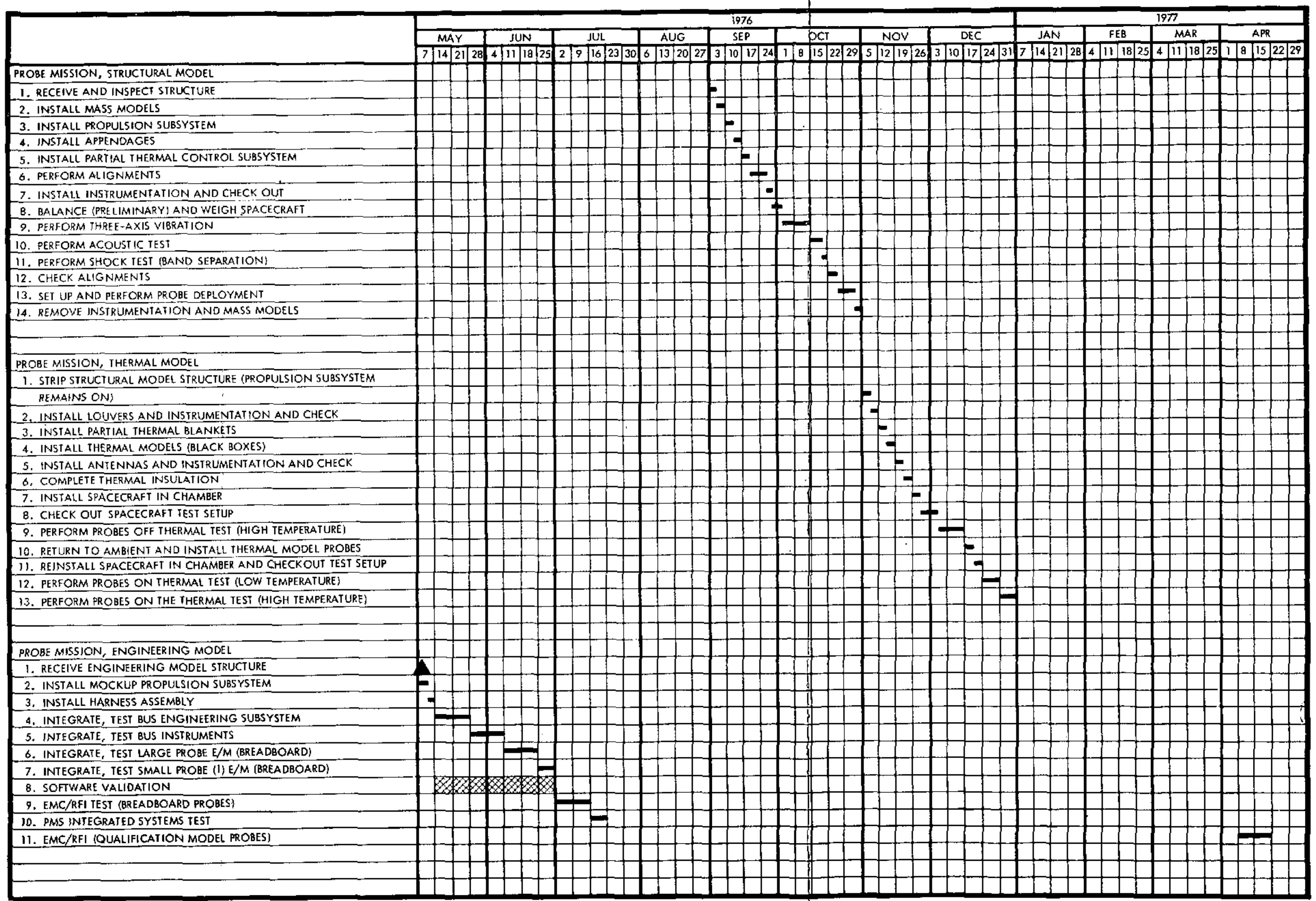


Figure 3-16. Probe Mission, Structural, Thermal, and Engineering Model Schedule

3.4.4 Qualitative Plan Comparisons

3.4.4.1 Test Philosophy Comparison of Plans

The advantages in progressing from plan A through plans B to C are clearly the reductions in redundant test activities; this factor is illustrated by the combined test objective table and spacecraft model chart for each plan (see Table 4-6).

Plan B, which combines engineering and prototype test objectives by expanded prototype testing, shows some improvement over the basic test program. Plan C retains the engineering model for electrical system functional related interface checks and software development and combines the prototype and flight acceptance objectives in the proto/flight spacecraft. The further reduction in test redundancy is evident.

All the candidate plans retain the structural and thermal test model as the primary source of both the mechanical dynamic response and thermal induced environmental data, and verification of the structure mechanical adequacy and analytic thermal computer model.

3.4.5 Quantitative Plan Comparisons

The reduction in test overlap from the traditional plan (plan A) to the more efficient proto flight program (plan C) results in significant cost savings. The cost savings arise from a number of areas:

- Hardware utilization
- Orbiter structure design and test efficiency
- Integration and test planning and implementation.

3.4.5.1 Hardware Utilization

Table 3-7 shows hardware utilization and identifiable cost savings for each plan. The significant differences show in the use of structures, electronic boxes, thermal control, and cable harnesses. In plans A and B, a single structure is used for the structural/thermal model and prototype testing (probe mission); the structure is then reconfigured and refurbished for the orbiter configuration and used for structural/thermal model and prototype testing (orbiter mission). New structures are manufactured for both the probe and orbiter flight spacecraft, making a total of three structures plus one major reconfiguration.

Table 3-6. Utilization of Spacecraft Models for Each Plan to Meet Test Objectives

TEST OBJECTIVES	SPACECRAFT MODELS									
	PLAN A				PLAN B			PLAN C		
	STM	ETM	PROTOTYPE	FLIGHT	STM	PROTOTYPE	FLIGHT	STM	ETM	PROTO/FLIGHT
DEMONSTRATE SPACECRAFT STRUCTURE ADEQUACY AT SPECIFIED (QUAL LEVEL) DYNAMIC ENVIRONMENTS	●				●			●		
DETERMINE SPACECRAFT STRUCTURE ADEQUACY UNDER EXPECTED LAUNCH BOOSTER LOADS	●				●			●		
PROVIDE DATA FOR ESTABLISHING ACOUSTIC, VIBRATION AND SHOCK INDUCED ENVIRONMENTS FOR ALL SUBSYSTEMS AND INSTRUMENTS	●				●			●		
VERIFY ANALYTIC COMPUTER THERMAL MODEL	●		●		●	●		●		●
VERIFY PROPER THERMAL INSULATION DESIGN AND FIT	●		●		●	●		●		●
PROVIDE DATA FOR THERMAL INDUCED ENVIRONMENTS FOR ALL SUBSYSTEMS UNDER ALL MISSION CONDITIONS	●		●		●	●		●		●
INTEGRATION AND TEST CREW EDUCATION AND ORIENTATION		●	●			●	●		●	●
INTEGRATION AND TEST PROCEDURE VERIFICATION		●	●			●			●	●
GROUND SUPPORT EQUIPMENT (HARDWARE AND SOFTWARE) OPERATION VERIFICATION		●	●			●			●	●
GROUND SUPPORT EQUIPMENT TO SPACECRAFT INTERFACE VERIFICATION	●	●	●		●	●		●	●	
SPACECRAFT SUBSYSTEM INTEGRATION AND OPERATION VERIFICATION		●	●	●		●	●		●	●
SPACECRAFT SUBSYSTEM TO SUBSYSTEM INTERFACE VERIFICATION		●	●	●		●	●		●	●
SCIENTIFIC INSTRUMENTS INTEGRATION AND OPERATION VERIFICATION		●	●	●		●	●		●	●
SCIENTIFIC INSTRUMENTS TO SUBSYSTEM INTERFACE VERIFICATION		●	●	●		●	●		●	●
VERIFICATION OF SPACECRAFT SUBSYSTEMS AND INSTRUMENT SYSTEM LEVEL OPERATIONAL VERIFICATION		●	●	●		●	●		●	●
REACTION CONTROL SUBSYSTEM INTEGRATION AND OPERATION PERFORMANCE TEST (PROPULSION TANKS, LINES, AND THRUSTER ASSEMBLIES)		●	●	●		●	●			●
APPENDAGE AND/OR PROBE DEPLOYMENT VERIFICATION	●		●	●	●	●	●	●		●
SPACECRAFT SUBSYSTEM AND INSTRUMENT SYSTEM LEVEL OPERATION AND SURVIVAL UNDER MECHANICAL ENVIRONMENTS			●	●		●	●			●
SPACECRAFT SUBSYSTEM AND INSTRUMENT SYSTEM LEVEL OPERATION UNDER SPACE SIMULATION ENVIRONMENTS			●	●		●	●			●
VERIFICATION OF SPACECRAFT MAGNETIC PROPERTIES				●			●			●
VERIFICATION OF SPACECRAFT PHYSICAL PROPERTIES (ALIGNMENTS, WEIGHT, CENTER OF GRAVITY, MOMENTS OF INERTIA)			●	●		●	●			●
SPARES READINESS (INTEGRATE SPARES ON ETM OR PROTO)			●			●			●	
GROUND STATION COMPATIBILITY VERIFICATION			●	●		●	●		●	●
LAUNCH VEHICLE COMPATIBILITY VERIFICATION			●	●		●	●	●		●
SPACECRAFT SUBSYSTEM AND INSTRUMENT HEALTH MONITORING AND CONFIGURATION THROUGH LAUNCH OPERATIONS TO LIFT OFF				●			●			●

Table 3-7. Hardware Utilization Comparisons for Alternative Plans

HARDWARE REQUIREMENTS	PLAN A	PLAN B	PLAN C
STRUCTURES	3*	3*	2
ELECTRONIC BOXES (SETS)	4	3	3
SOLAR ARRAY SETS	2	2	2
SCIENTIFIC INSTRUMENTS	3	2	2
PROPULSION SUBSYSTEMS	3	3	3
THERMAL CONTROL	7	7	5
CABLE HARNESSES	5	4	4
COST REDUCTIONS (COMPARED TO PLAN A)			
HARDWARE		$480 000	$970 000
DESIGN COST (STRUCTURE)**		-	60 000
TOTAL		$480 000	$1 030 000

*STRUCTURE RECONFIGURATION FROM PROBE TO ORBITER MISSION.

**PLAN C USES ONE STRUCTURE FOR EACH MISSION TEST PROGRAM. THIS ALLOWS EARLY DESIGN AND TEST OF THE ORBITER STRUCTURE IN PARALLEL WITH PROBE BUS TEST PLAN AND RESULTS IN EFFICIENCY COST SAVINGS IN STRUCTURE DESIGN AND TEST.

In plan C, a total of two structures are manufactured; one for each mission (probe and orbiter). Each structure is used for the structural/thermal testing and then refurbished for use as the flight structure.

The manufacturing cost for each structure is estimated at $190, 000. The reconfiguration cost between the probe and orbiter configuration is estimated at $100, 000.

Because the plan A start-up is earlier (compared to plans B and C), a set of electronic boxes will be required early to support the plan A engineering model spacecraft. These electronic boxes will probably be too early in the program to include flight approval and thus have no multiple use. The additional cost of providing the engineering model boxes is estimated at $400, 000. The first spacecraft integrated in plans B or C will use prototype or qualification model electronics, which may be refurbished later as flight spares. The differences in quantities of thermal control subsystems and cable harnesses required for the various plans result directly from the number of spacecraft models used and, in the case of thermal control, the flight spares requirements.

The thermal control subsystem is estimated at $100, 000 per copy and the cable harness is estimated at $40, 000 per copy.

3.4.5.2 Integration and Test Planning and Implementation

Plans B and C are 4 and 6 months shorter than plan A. This results in reductions of management and crew levels of effort estimated at approximately $200, 000 total (see Table 3-8).

Table 3-8. Cost and Schedule Comparisons for Alternative Plans

		PLAN A	PLAN B	PLAN C
SCHEDULE (MONTHS)				
PROBE MISSION		19	17	17
ORBITER MISSION		17	15	15
		36	32	32
COST REDUCTIONS (COMPARED TO PLAN A)				
TEST PLANNING AND IMPLEMENTATION		-	$180 000	$270 000
COST SAVINGS GENERIC TO ALL PLANS, $000				
PROCEDURES	60			
SOFTWARE	250			
PREFLIGHT CALIBRATIONS	110			
INTERFACE TESTING	20			
	440			

The proto/flight testing concept in plan C leads to further cost savings in the reduction of environmental testing by combining elements of prototype qualification and flight acceptance testing instead of separate test models for qualification and acceptance tests.

By combining elements of the qualification and acceptance testing, it is estimated that cost savings of $45, 000 in test facility support (operating thermal vacuum chambers, lamps, and vibrators) will result from both probe and orbiter mission test programs.

3.4.5.3 Prototype/Flight - The Preferred Plan

The proto/flight concept exposes the spacecraft system to acceptance-level vibration, shocks, and acoustics rather than qualification levels. The supporting rationale is:

- All bus/orbiter subsystems and probes designs will have been qualified at the unit level
- All bus/orbiter subsystem and probe units have been acceptance tested
- The only remaining unit not tested to qualification levels is the harness and the thermal blankets.

Therefore, acceptance level mechanical environments are sufficient to verify the integrity of the harness and insulation installation.

The proto/flight concept does provide for two thermal vacuum tests. The first test uses the updated thermal design (based on results of the thermal model test) and flight hardware. This test provides a final evaluation of the thermal design and also provides an opportunity to evaluate the performance of the other subsystems and science. The second thermal vacuum test verifies the final thermal design and the spacecraft/science systems.

TRW believes that this test concept provides adequate confidence in the performance of the spacecraft systems at the lowest cost. It is the preferred plan.

3.4.6 Stacked versus Series Schedule Tradeoffs (Plan C Only)

TRW has evaluated two alternate system integration and test approaches for the orbiter and probe mission spacecraft, based on plan C.

1) Series Schedule is based on the orbiter mission structural/thermal model, engineering model, and the initial series of electrical integration and systems test activities of the proto/flight spacecraft to be performed prior to the start of the probe mission systems test activities. Figure 3-17 shows the major milestone tasks on this series test schedule.

2) Stacked Schedule is based upon parallel testing of the orbiter mission engineering model, and proto/flight spacecraft in a more condensed schedule time than the series schedule previously mentioned. Figure 3-18 shows the major milestone tasks on this stacked test schedule.

3.4.6.1 Series Schedule

The series schedule has the following advantages and disadvantages.

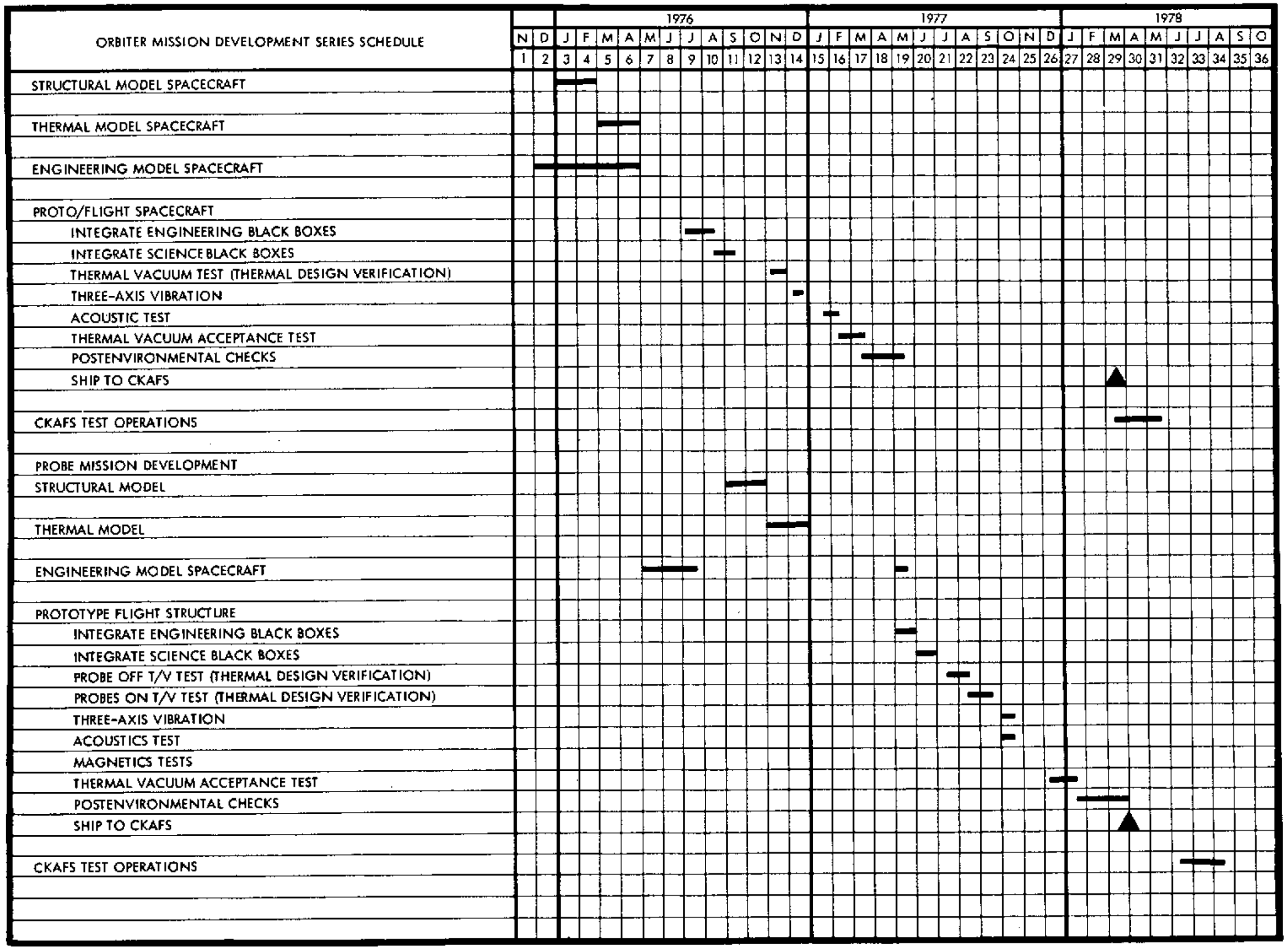


Figure 3-17. Orbiter Mission Development Series Schedule

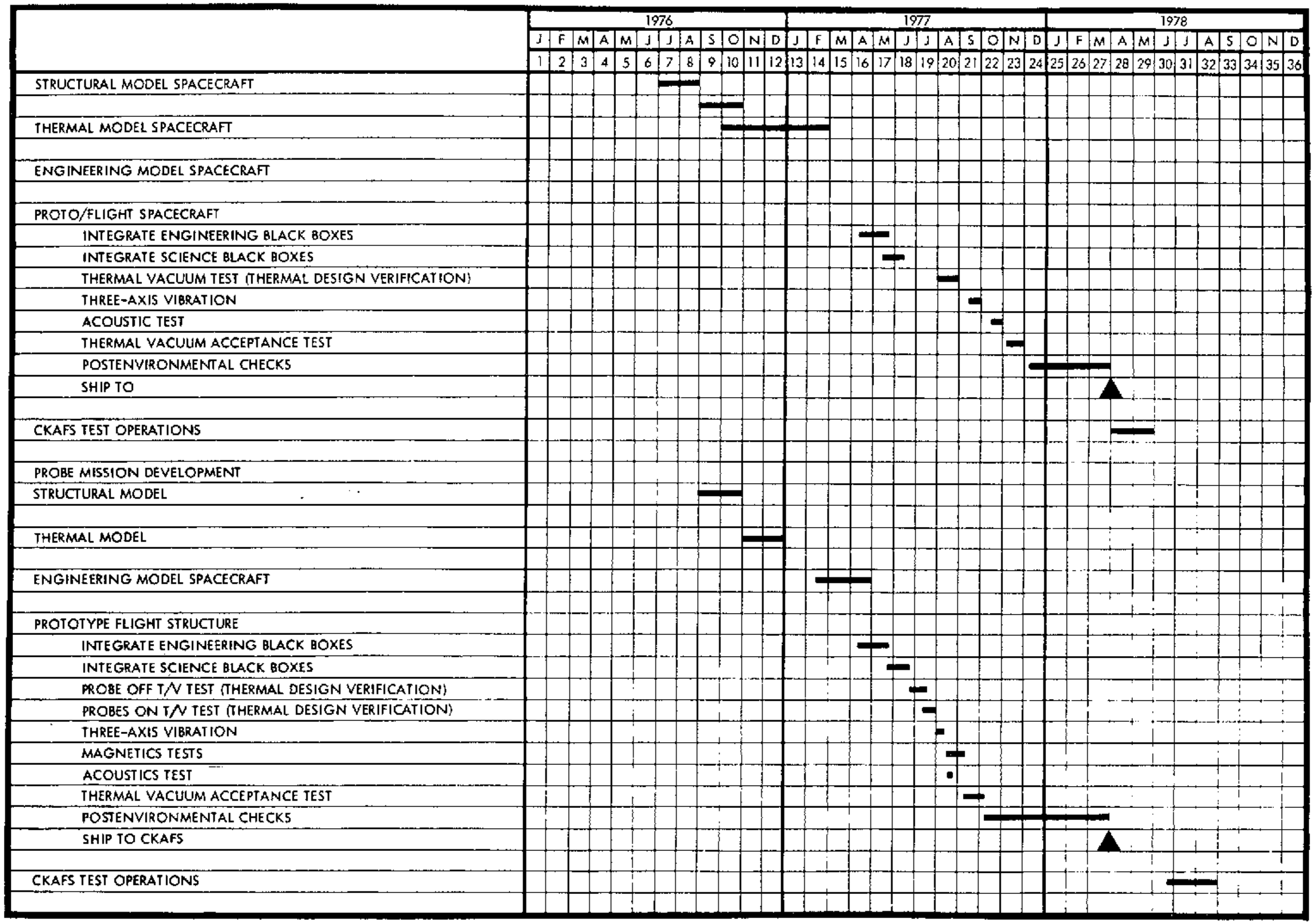


Figure 3-18. Orbiter Mission Development Stacked Schedule

Advantages

- Early start nonparallel testing assures maximum recovery time in the event of problems, i.e., minimum schedule/program risk compared to stacked schedule.

- Requires the use of only one EGSE/system test station. Any parallel orbiter mission and probe mission test activities required due to hardware or test induced schedule problems could be worked on a second shift basis. Requires procurement of only one set of ADPE for the EGSE/STS.

- Allows time for periodic electrical spot checks on orbiter mission proto/flight spacecraft during storage, while the probe mission proto/flight spacecraft is undergoing mechanical system-level tests.

Disadvantages

- Early startup requires test personnel on program earlier than on stacked schedule.

- Allows less time for correction of black-box problems observed during fabrication, acceptance development program, i.e., orbiter black boxes are required earlier in the program than on the stacked schedule. However, workarounds, using engineering model units, are available to maintain schedule.

3.4.6.2 Stacked Schedule

The stacked schedule has the following advantages and disadvantages:

Advantages

- Late startup time allows maximum time for resolution of black box or probe development problems observed during fabrication qualification/acceptance operations.

- Less total manmonths of test personnel support are involved with parallel/stacked schedule operations. A full two test crews would not be required to support the systems test activities on the two spacecrafts.

Disadvantages

- Maximum schedule/program risk due to limited recovery time to resolve problems encountered during systems level tests.

- Requires procurement, and subsequent maintenance of second ADPE for EGSE/system test station.

- Requires refurbishment of entire second EGSE test station used on Pioneers 10 and 11. This includes fabrication of two of each drawer or peripheral EGSE equipment being developed for the Pioneer Venus program.

- Requires phasing of test operations to allow for thermal vacuum testing on the two spacecraft to be performed in series. Additional fixturing and test personnel support would be required to support parallel thermal vacuum tests. This would put excessive strain on human and physical resources available to the program.

3.4.7 Systems Magnetic Tests Tradeoff (Orbiter Mission)

3.4.7.1 Background

Stray fields and static perm/deperm magnetic tests were performed on the Pioneer 10 and 11 spacecraft at the TRW Malibu Test Site to prove the component of magnetic field in the spin-axis direction at the deployed magnetometer sensor that was induced by the spacecraft (less science packages) did not exceed 0.03 gamma stray fields, and 0.04 gamma remanence after demagnetization.

The Version IV science payload redirection indicates the equivalent stray field and remanence depermed fields for the Venus orbiter spacecraft design will be approximately 0.5 gamma. This comparable higher allowable magnetic field will allow a reduction in the complexity (and associated costs) of the test setup and test implementation of the systems level magnetic tests on the orbiter mission.

3.4.7.2 Test Description

A final spacecraft test is required because of the deletion of many lower level tests as compared to the Pioneer 10 and 11 programs. It also gives a better measure of the predicted spacecraft field after launch since it is left in a known demagnetized state.

An integrated spacecraft test will be performed in a magnetically clean area where most extraneous activity is controllable and during times when uncontrollable activity is at a minimum. Techniques developed for the Pioneer Jupiter spacecraft program will be used. Higher level signals are measured at various closer-in distances, as well as at the actual magnetometer sensor position, to circumvent ambient noise level limitations.

Our prior experience in testing Pioneers 10 and 11 has shown that a coil system nulling the geomagnetic field is essential in eliminating induced moments in the spacecraft remanent field test. We have determined that nulling to 10 percent over the spacecraft volume would be adequate for the Pioneer Venus orbiter and a single tilted, 7.3-meter (24-foot) square coil will be used, as opposed to the more complex and costly three-axis coil system used on the Pioneer 10 and 11 test program. The proposed tilted configuration (shown in Figure 3-19) permits a ground level translation of the spacecraft without burying a part of the coil.

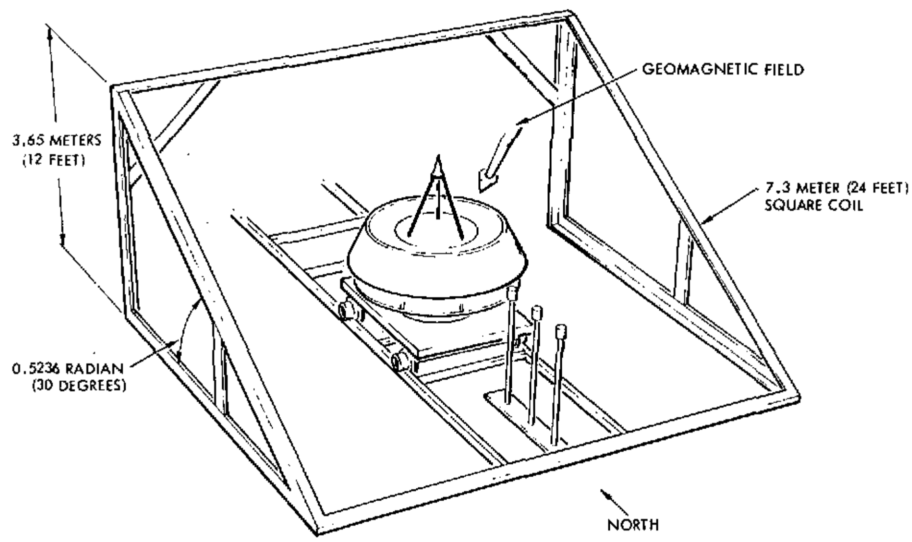


Figure 3-19. Proposed Tilted Square Coil to Buck Out Geomagnetic Field

3.4.7.3 Magnetic Control-System Level

Monitoring the spacecraft magnetic environment after the last demagnetization and avoiding the use of magnetic tools during all spacecraft test periods/exposures prior to launch is essential for magnetic integrity. GFE flux monitors, as used on Pioneers 10 and 11, will be mounted on the spacecraft at the appropriate time and be checked periodically prior to launch. Plans will be made for recalibration, demagnetization, or recompensation up to the latest possible time on the schedule in the event of accidental exposure to high-intensity magnetic environment.

3.5 LAUNCH OPERATIONS

This study effort was performed to identify differences in scheduling/planning, facility support requirements, and personnel support requirements for orbiter mission and probe mission operations at CKAFS (dual 1978) launch in the Thor/Delta and the Atlas/Centaur launch vehicle configurations.

3.5.1 Test Facilities

Initial prelaunch checkout of the orbiter and the probe missions would be performed in the same hangar checkout facility regardless of the launch vehicle being employed. Hangar AO was used on Pioneers 10 and 11 and incorporates extensive modifications to facility power, STS test area, etc., that would not require further modifications for Pioneer Venus usage. Prelaunch checkout operations could be performed in Hangars AE and AO, assuming facility power, air-conditioning, and area requirements could be satisfied. The following facilities would be involved in prelaunch test operations on the two launch vehicles.

Thor/Delta

- Hangar (AO)
- Solid propellant storage area (orbiter mission only)
- Delta spin test facility
- Launch complex 17

Atlas/Centaur

- Hangar (AO)
- Solid propellant storage area (orbiter mission only)
- Explosive safe area (PLB, and S A buildings)
- Launch complex 36

In general, major facility requirements in all these areas can be broken down as follows

Environmental control: 25 ± 1.5°C (75 ± 5°F)
Relative humidity: ≤ 50 percent

Crane: 9.07-ton (10-ton) maximum
Hook height above floor level 9.14-meter (30-foot) minimum
Vernier lift control

Spacecraft work area: 139.15 sq meter (1500 sq ft) minimum (except on stand)

3.5.2 Test Planning/Schedules

Figures 3-20 and 3-21 show a preliminary sequence/schedule for the orbiter and probe mission prelaunch test activities with the Thor/Delta launch vehicle. Figures 3-22 and 3-23 show the sequence/schedule for the orbiter and probe mission prelaunch test activities with the Atlas/Centaur launch vehicle.

No significant advantages or disadvantages were identified in performing prelaunch test operations in either the Thor/Celta or Atlas/Centaur launch vehicle configurations. Prelaunch test operations performed on Pad 17 (Thor/Delta) are somewhat easier to accomplish during the initial on-stand activities because the fairing is not installed until 1 to 2 days prior to launch. However, payload air-contiditioning/cleanliness is more difficult to maintain in the fairing-off configuration.

For the Thor/Delta launch vehicle, spacecraft test operations in the Delta spin test facility are limited to hydrazine loading, TCA hot firing, and spacecraft weighing (plus orbiter mission SRM mating). The spacecraft is shipped to complex 17 and mated to the Delta interstage adapter previously erected.

For the Atlas/Centaur launch vehicle, spacecraft test operations in the ESA complex include hydrazine loading, TCA hot firing, weighing (orbiter mission SRM mating), mating to Centaur interstage adapter, fairing encapsulation. The mated, encapsulated spacecraft is then shipped to complex 36 for mating to the Atlas/Centaur launch vehicle.

A review of the schedules and the level of orbiter and probe mission test personnel support required indicates there are no significant technical or cost advantages in either the Atlas/Centaur or the Thor/Delta configuration, as far as TRW launch support activities are concerned. The Thor/Delta simulated flight test is normally performed approximately 10 working days prior to launch before the payload is mated. The

Thor/Delta countdown is now a 4-day exercise. The Atlas/Centaur FED and CERT tests are performed after payload mating. The Atlas/Centaur countdown is usually 2 days.

The EGSE/STS ground control console would be located on the spacecraft checkout level of the missile service tower on both pads 17 and 36. No interface problems were identified in the use of this equipment at this location.

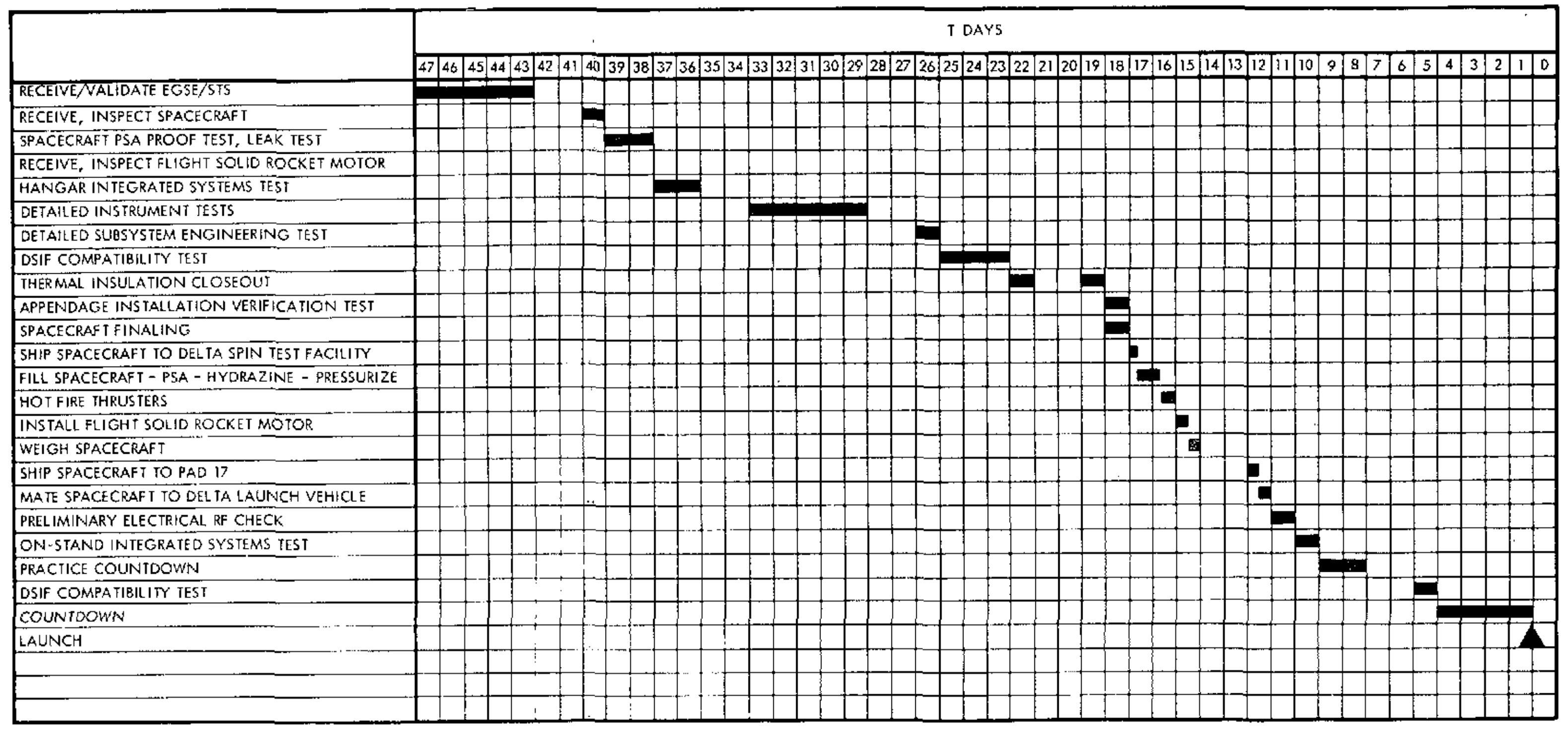


Figure 3-20. Thor/Delta Launch Vehicle Proto/Flight Orbiter Mission Operations (CKAFS)

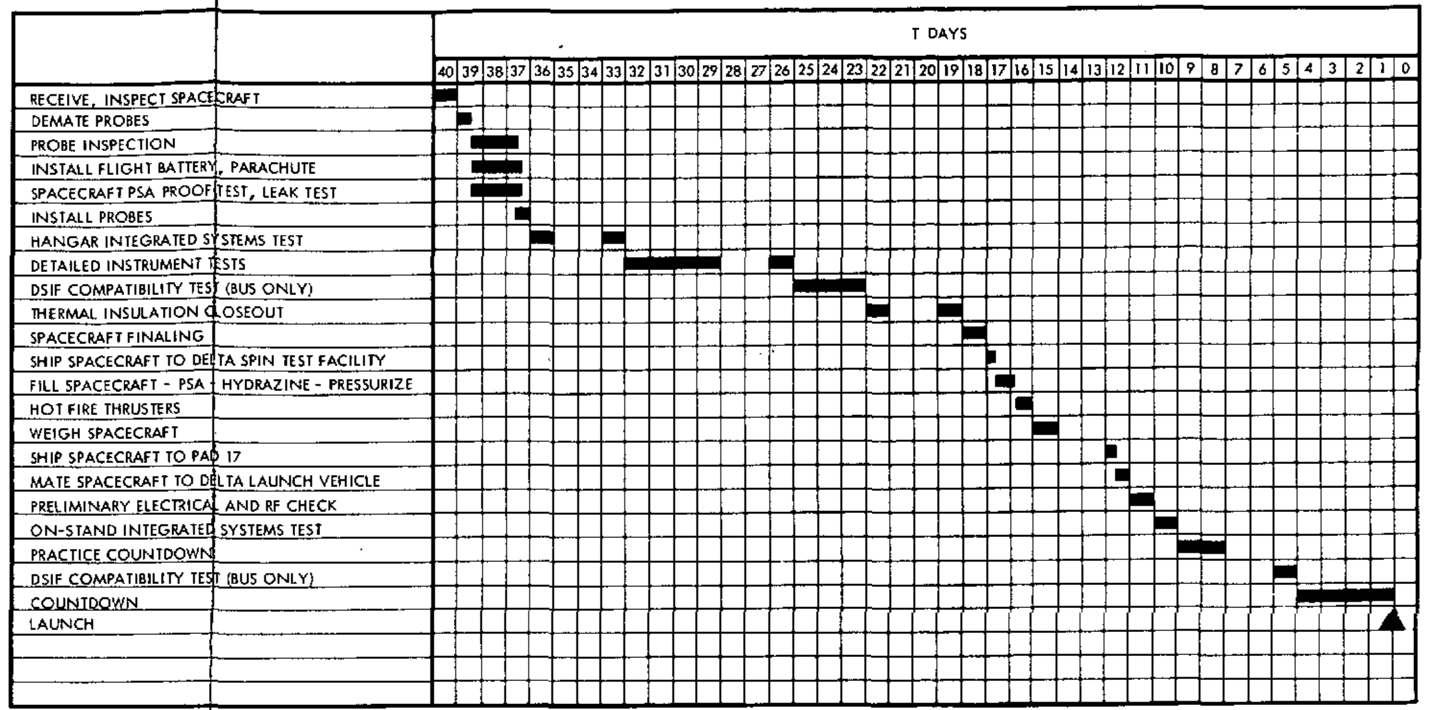


Figure 3-21. Thor/Delta Launch Vehicle Proto/Flight Probe Mission Operations (CKAFS)

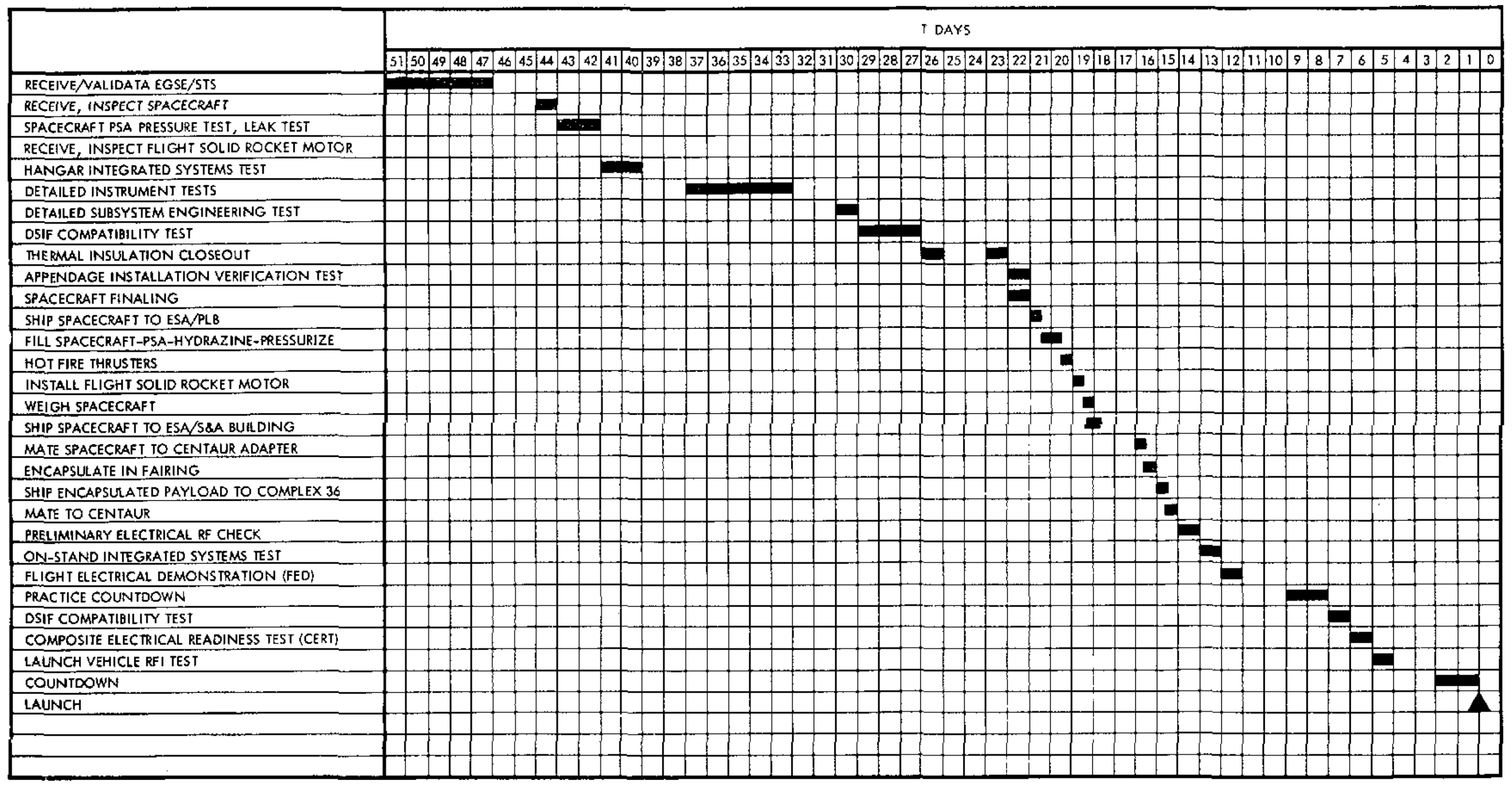


Figure 3-22. Atlas/Centaur Launch Vehicle Proto/Flight Orbiter Mission Operations (CKAFS)

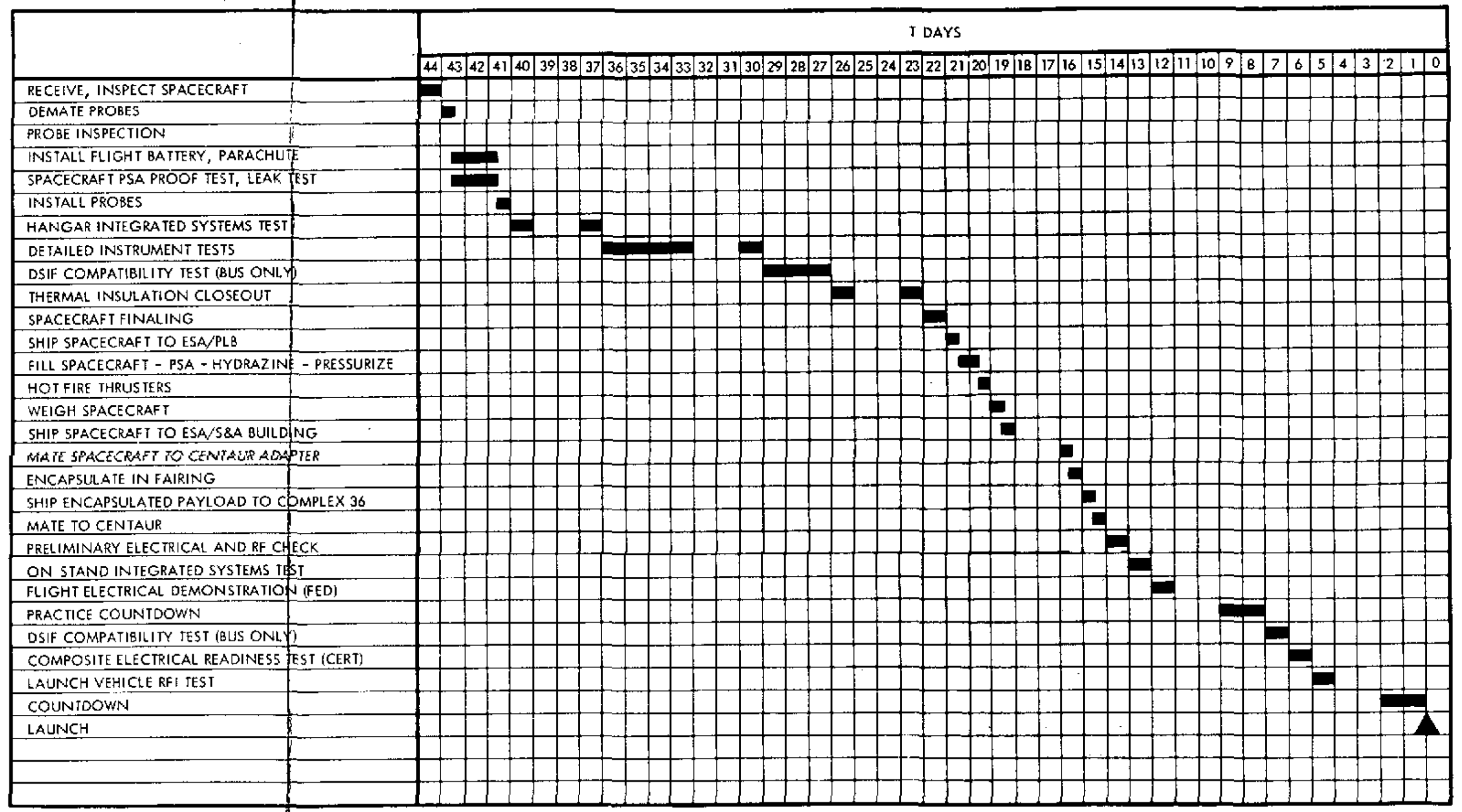


Figure 3-23. Atlas/Centaur Launch Vehicle Proto/Flight Probe Mission Operations (CKAFS)

FOLDOUT FRAME 1

FOLDOUT FRAME 2

4. SYSTEM TEST ELECTRICAL GROUND SUPPORT EQUIPMENT

An analysis of the factory through launch operations has been performed to identify the electrical ground support equipment (EGSE) required to support the manufacturing, integration, test, and launch of the orbiter mission and probe mission spacecraft. The study has included all levels of hardware from the probe system level through the fully integrated flight spacecraft in the launch configuration. Test locations considered included Martin Marietta Denver, Martin Marietta Orlando, TRW, and the launch site.

Emphasis was given to the use of Pioneer 10 and 11 EGSE and to the commonality between Pioneers 10 and 11 and Pioneer Venus. Another major consideration was the EGSE for the probes since they constitute a very important portion of the Pioneer Venus program and represent the most significant change from the Pioneer 10 and 11 program. A major portion of the study was devoted to a computer tradeoff study to determine the automatic data processing equipment (ADPE) configuration since it constitutes a significant part of the system test set (STS).

4.1 STUDY RESULTS, CONCLUSIONS, AND RECOMMENDATIONS

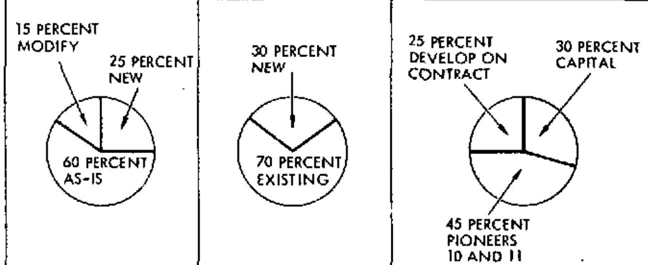
Economic factors were emphasized in this study. The four most promising methods for reducing the STS program costs were determined to be:

- Use of existing equipment
- Joint use of EGSE hardware and software for probe and spacecraft testing
- Use of existing software
- Multiple usage of probe test equipment for testing subsystem and system.

Figure 4-1 is included to give a gross indication of the Pioneers 10 and 11, TRW and Martin Marietta capital equipment that is available for use on the Pioneer Venus program. The sources of the Pioneer Venus EGSE (excluding the ADPE) will be as follows:

		DESIGN STATUS			EQUIPMENT STATUS		EQUIPMENT SOURCE		
		AS-IS	MODIFY	NEW	EXISTING	NEW	CAPITAL	PIONEERS 10 AND 11	DEVELOP ON CONTRACT
TRW	ADPE *	100			TBD			TBD	TBD
TRW	GROUND CONTROL CONSOLE	20	40	40	50	50	20	15	65
TRW	TAPE RECORDER CONSOLE	95	5		100			95	5
TRW	RF CONSOLE	90	10		100			85	10
TRW	TELEMETRY DATA CONSOLE	70	30		100			30	20
TRW	TEST CONDUCTOR CONSOLE	100			100			95	
TRW	PERIPHERAL EQUIPMENT	20	30	50	50	50		20	80
TRW	DATA FORMAT GENERATOR	200			100			100	
TRW	SPACECRAFT SIMULATOR	75	15	10	90	10		75	25
MMC	PROBE TEST EQUIPMENT			100		100			100
MMC	BUS/PROBE INTERFACE SIMULATOR			100		100			100
MMC	PROBE PYRO SIMULATOR			100		100			100
MMC	PROBE PYRO TESTER	100			100		100		
MMC	PROBE DPE	100			100		100		

APPLICABLE WEIGHTS FROM ABOVE APPLIED TO DERIVE THE PERCENTAGES



*ADPE IS NOT INCLUDED IN THE PERCENTAGES

Figure 4-1. EGSE Summary Status

- 45 percent from Pioneers 10 and 11
- 30 percent from TRW and Martin Marietta capital
- 25 percent Pioneer Venus.

These percentages would be further enhanced by the use of a Sigma 5 ADPE GFE.

The Chart in Figure 4-2 indicates the percentage of EGSE hardware and software that can be shared for probe testing and spacecraft testing, as estimated by the experienced Pioneer 10 and 11 personnel. The large ratio of shared hardware and software result from:

- Design of probe test equipment to interface with both the Martin Marietta ADPE and the TRW STS ADPE
- Commonality in probe and spacecraft telemetry formats
- Design software using Fortran to enable use in either Martin Marietta or TRW ADPE.

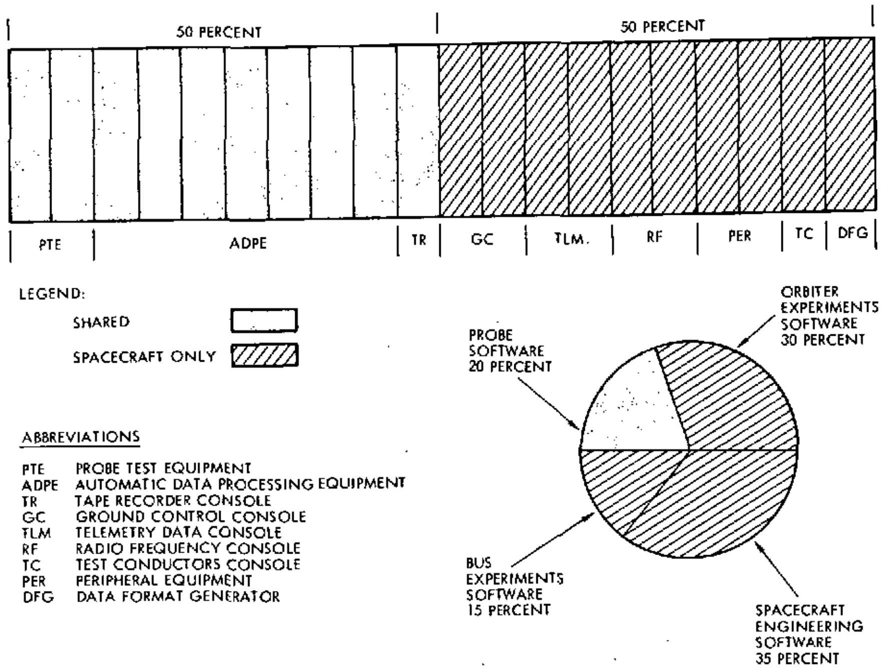


Figure 4-2. Joint Use of EGSE and Software for Probe and Spacecraft Test at TRW and CKAFS

The advantages of Martin Marietta/TRW shared hardware and software for probe testing include cost savings in both hardware and software and food correlation of test data at all test configurations.

The Pioneer 10 and 11 Sigma 5 is recommended as the configuration for the STS ADPE. The chart in Figure 4-3 illustrates the software savings by using the Sigma 5; the rationale for the indicated percentages is given in Section 4.4. Approximately 30 percent of the STS operational software can be used from Pioneers 10 and 11.

The use of the Pioneer 10 and 11 EGSE is a sound technical approach as well as an economical cost approach since this equipment has demonstrated a high degree of reliability. The new items of EGSE for Pioneer Venus will be based on existing technology.

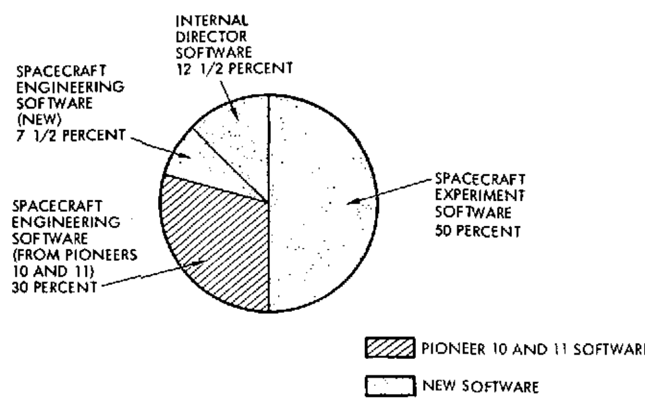


Figure 4-3. Spacecraft Software Sigma 5 Configuration

4.2 PROGRAM REQUIREMENTS

The general program requirements that have a strong influence on the EGSE include test configurations, test locations, and test schedules.

4.2.1 Test Configuration

The test configurations were established during the preparation of the test plan as the integration flows were optimized. The system test configurations for the two missions are:

	Spacecraft	
	Probe	Orbiter
Small probe	X	
Large probe	X	
Experiments	X	X
Bus	X	
Spacecraft	X	X

The small probe and large probe tests will include both open and sealed tests. The open tests will consist of functional tests performed as the components and subassemblies of the probes are integrated into a fully operating probe system. The sealed tests will include functional performance and environmental tests of the probes prior to their being integrated onto the bus.

The experiment tests will include the functional bench tests performed on the experiments as they are received from the various principal investigators and prior to their being integrated onto the bus. Similar tests were performed on the Pioneer 10 and 11 experiments.

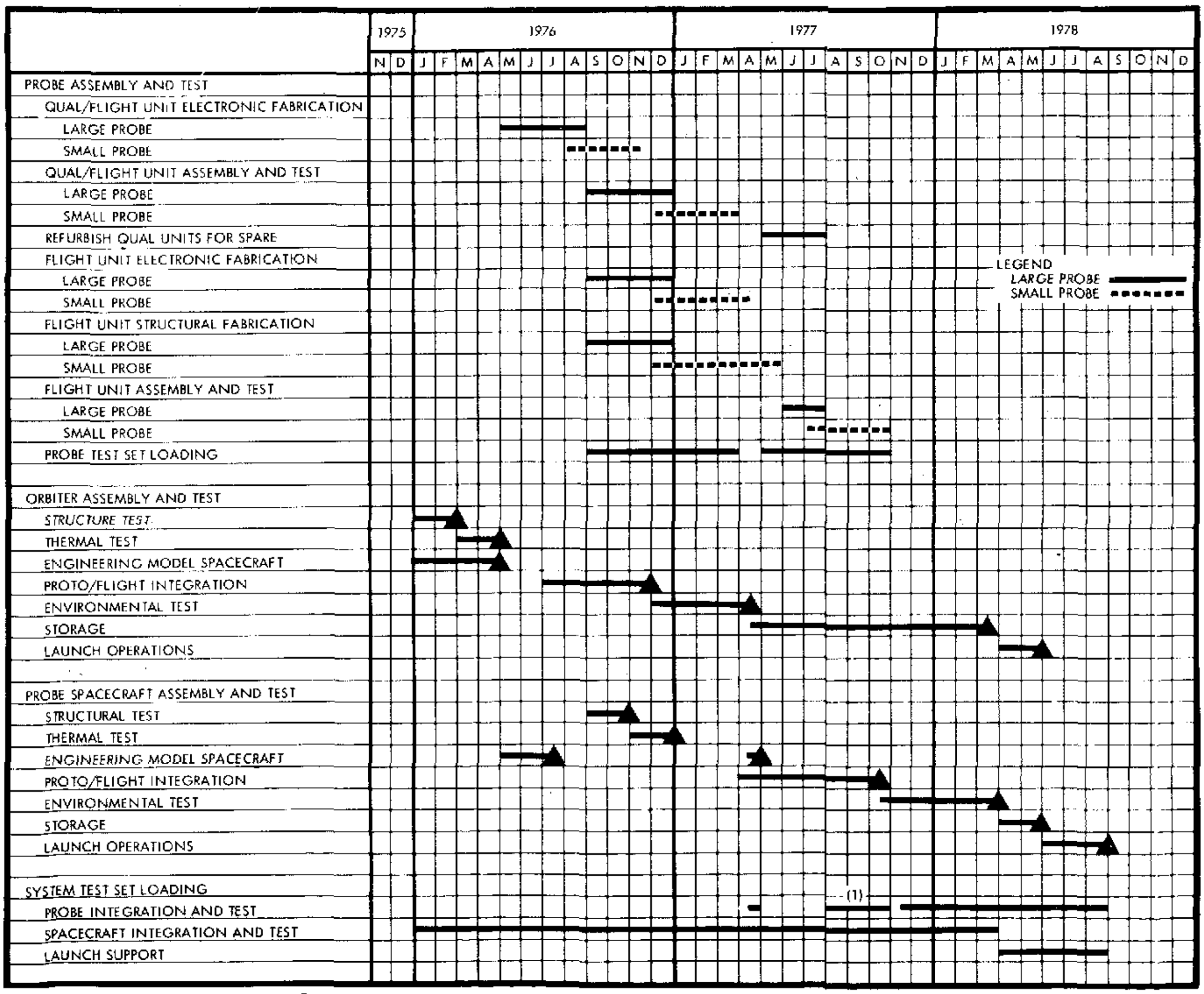
The spacecraft tests will initially include the functional tests performed on either the probe bus or the orbiter during their integration cycle prior to integration of experiments or probes.

The bus and spacecraft testing at TRW and the launch site will include much commonality of functional tests. The same system test set will be used to support the testing at these two locations.

4.2.2 Test Schedules and EGSE Loading

The test schedules have a direct bearing on the amount of EGSE required on the program especially since various items of EGSE are candidates for multiple usage. The test schedule in Figure 4-4 shows the time phasing of the various test levels at each of the test locations and the loading diagram for the EGSE required to support this schedule.

Spacecraft tests showing integration of the probes and experiments will include the tests performed on either the probe or the orbiter spacecraft. These tests will include functional performance and environmental tests.



NOTE (1): PROBE QUAL TEST DOES NOT REQUIRE PROBE TEST SET. USES SPACECRAFT TELEMETRY OR HARDLINE VIA TELEMETRY CONSOLE.

Figure 4-4. Integration and Test, EGSE Loading Schecule

4.2.3 Test Locations

Table 4-1 lists the test locations for the various system test configurations.

Table 4-1. Test Locations

TEST LEVEL	TEST LOCATIONS: MMC DENVER	MMC ORLANDO	TRW	LAUNCH SITE
SMALL PROBE (ALONG)	X	X	X	X (1)
LARGE PROBE (ALONG)	X	X	X	X (1)
EXPERIMENTS			X	X (1)
BUS			X	
PROBE SPACECRAFT			X	X
ORBITER SPACECRAFT			X	X

(1) IF REQUIRED

The small probe and large probe testing at Martin Marietta Denver, Martin Marietta Orlando, and TRW will include functional tests with much commonality. The probe system test set will be designed so that a small portion of it can be shipped from Martin Marietta Denver to each of the other locations to support testing at that site. The multiple use of this equipment is shown graphically in Figure 4-5. This approach has several attractive technical and cost advantages, i.e.:

- Common test procedures
- Good correlation of test data
- Minimum amount of test equipment
- Minimum shipping and handling for support equipment.

4.3 EGSE DESCRIPTION

The EGSE required for system testing and launch of the probe and orbiter mission spacecraft is divided into groups according to use as follows:

- System test set — supports the integration, test, and launch of the probes and spacecraft
- Probe EGSE — supports the integration and test of the probes prior to integration on the probe spacecraft (note some of the probe EGSE is also used in conjunction with the system test set to support probe testing after their integration onto the probe spacecraft)
- Spacecraft interface simulator — supports experiment receiving tests at TRW.

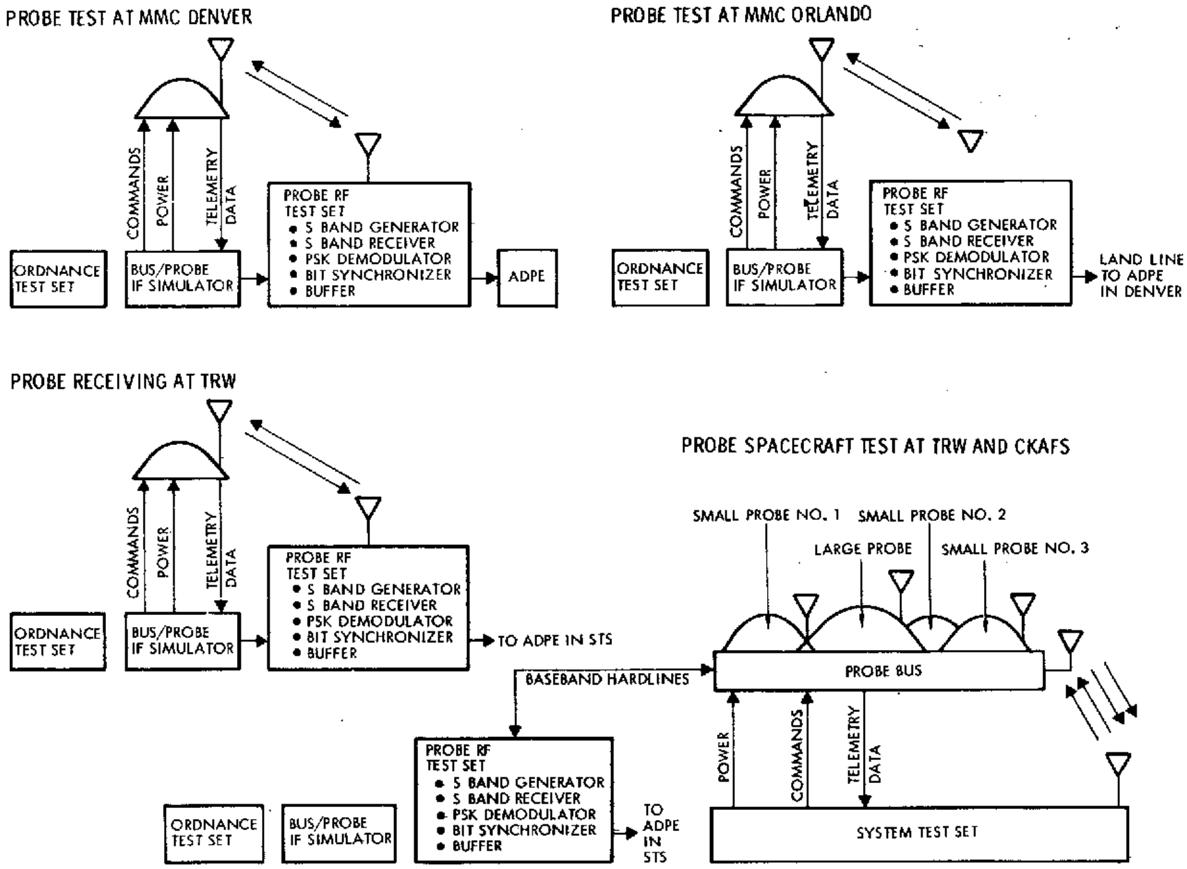


Figure 4-5. Multiple Use of EGSE

A summary of the EGSE that has been identified during the Pioneer Venus study is listed in Section 4.6. A description of the EGSE requirements and equipment implementation is described in the following sections.

4.3.1 System Test Set

The STS provides the necessary equipment support for spacecraft system testing during the integration and test program and the launch operations of the Pioneer Venus spacecraft. The objective of the STS is to support the integration and test of the experiments, probes, and electronic subsystems of the spacecraft and perform spacecraft integrated systems, testing to the depth necessary to develop the required prelaunch performance confidence. The same set is to be used in-plant and at the launch site, using procedures common throughout the integration and test cycle. Baseline performance data will therefore be developed for continuous systems evaluation.

The STS performance requirements are:

- Supply electrical power to the spacecraft either in the external or solar array simulated modes
- Hardline monitor critical power subsystem parameters
- Decommutate real time telemetry data from bus for entry to the ADPE
- Continuously process, monitor, and limit check the real-time telemetry
- Demodulate probe telemetry data and format for entry into ADPE
- Provide two-way telemetry and command RF link with orbiter and probe mission spacecraft
- Format uplink baseband with command tones
- Format digital command data and generate command sequences
- Monitor ordnance firing circuits to assure all-fire current values during ordnance firing and less than no-fire current values at all other times
- Provide for installation tests of all ordnance devices and verify absence of stray voltages
- Simulate and stimulate attitude control subsystem sensors and monitor control subsystem responses
- Record and play back all telemetry data for off-line evaluation
- Provide battery charging and safe and arm monitoring during countdown
- Provide for radiated communications link between launch complex and hangar
- Provide for STS validation and fault isolation.

The Pioneer Venus spacecraft STS consists of the consoles and assemblies listed below:

- RF console
- Recorder console
- Ground control console
- Telemetry data console and data format generator
- Test conductor's console

- EGSE peripheral equipment
- ADPE
- Launch site data communications equipment
- Probe RF test set.

The block diagram in Figure 4-6 shows the interrelationships of the major items. Descriptions of the console and assemblies, except for the RF probe test set, are provided in the following paragraphs. The probe console is also used at Martin Marietta and is described in Section 4.3.2, Probe EGSE.

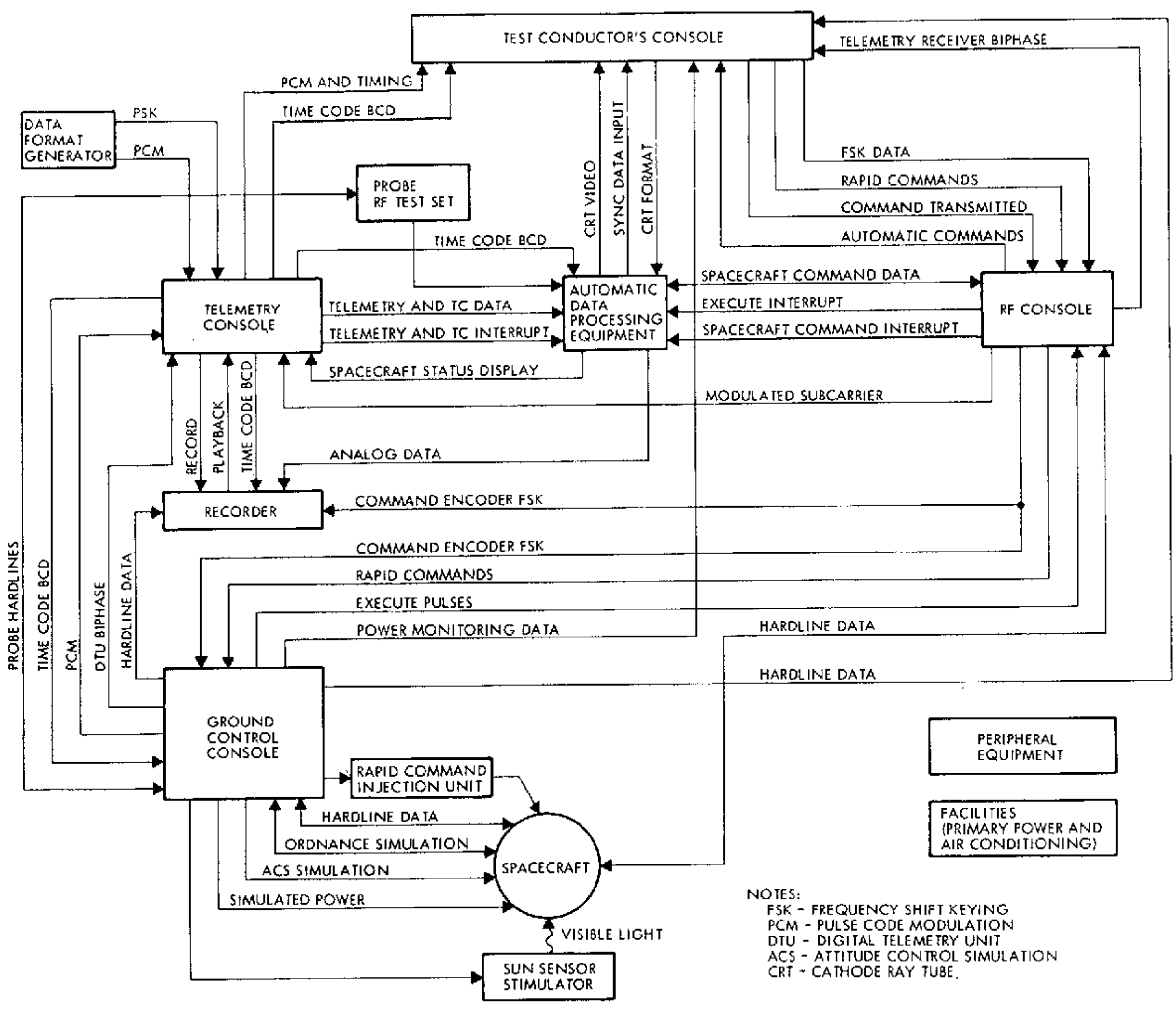


Figure 4-6. EGSE System Test Set Block Diagram

C-2

4.3.1.1 RF Console

The RF console provides the primary STS interface with the spacecraft. It transmits commands over an RF link to the spacecraft bus and receives and demodulates the downlink telemetry subcarrier for further processing and data extraction at the telemetry console. The functional requirements are:

- Receive downlink signals from the spacecraft bus
- Generate uplink signals to the spacecraft bus
- Generate command codes to modulate uplink transmitter
- Measure RF power, uplink, and downlink modulation index, RF frequency, and frequency stability
- Provide RF bypass link for injection of commands into the CDU and DDU
- Display time of day.

The RF console shown in Figure 4-7 contains the command transmitter, command encoder, telemetry receiver, encoder selector, ramp generator, command transmitter power supply, console power control, blowers, and frequency counter.

Since the Pioneer Venus spacecraft may be assigned different carrier frequency channels, it may be necessary to modify the command transmitter and telemetry receiver accordingly.

4.3.1.2 Recorder Console

The console will provide the means for direct-write analog recording, magnetic tape recording

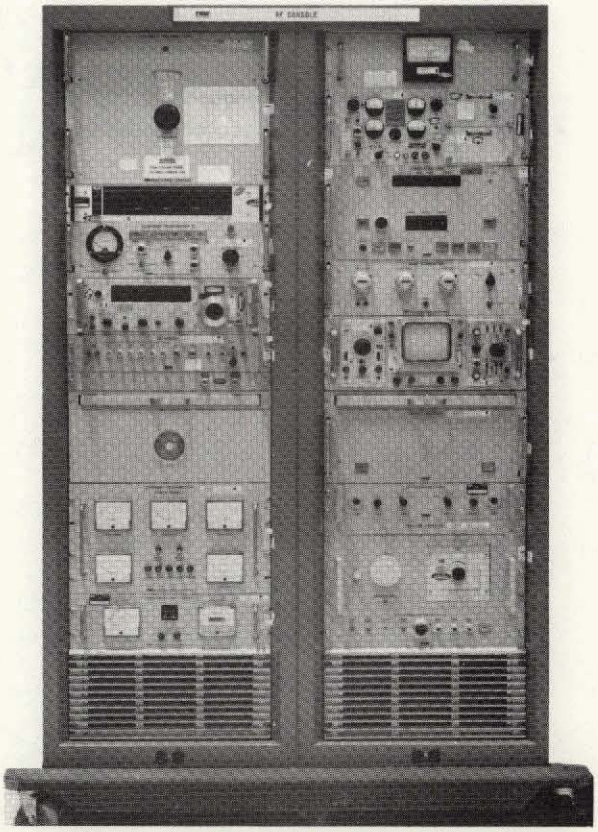


Figure 4-7. RF Console

and magnetic tape playback. The functional requirements are to provide:

- An 8-channel analog direct-write strip chart recording capability
- A 7-channel analog or digital recording capability in the form of an Ampex 1260 magnetic tape recorder
- Patching facilities enabling STS or spacecraft data to be patched to either recorder or other STS consoles.

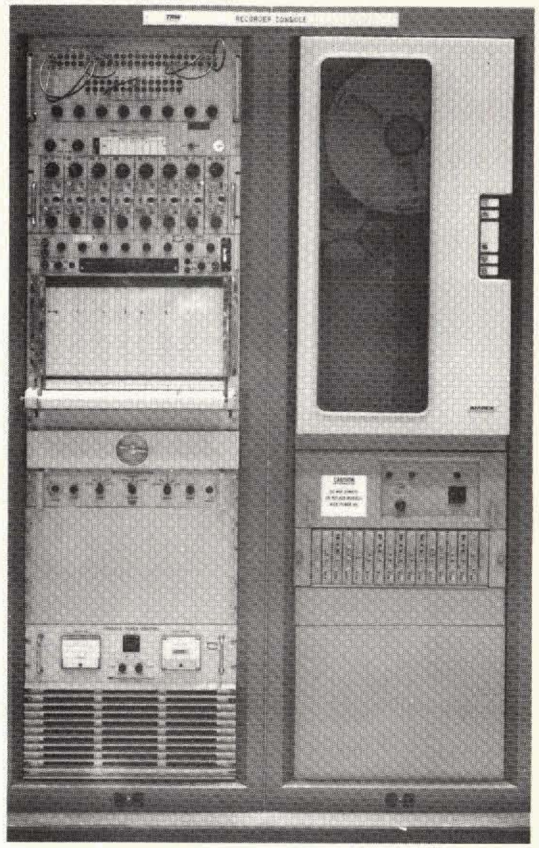


Figure 4-8. Recorder Console

The recorder console shown in Figure 4-8 contains a Sanborn 358-100A strip chart recorder, an Ampex 1260 magnetic tape recorder, an instrumentation patch panel, intercom, and power control assemblies.

Only the instrument patch panel will require modification due to different parameters being monitored and recorded for the Pioneer Venus program.

4.3.1.3 Ground Control Console

The ground control console is the primary electrical hardline interface unit for test and checkout of the spacecraft. It is capable of supporting three spacecraft subsystems: power, ordnance, and ACS. In addition, the unit serves as a monitoring, buffering, and distribution point for miscellaneous test signals. The functional requirements are listed in Table 4-2.

The ground control console shown in Figure 4-9 contains the following items:

- Test signal interface unit
- ACS simulation and control
- Intercommunication

Table 4-2. Ground Control Console Functional Requirements

FUNCTION	REQUIREMENT
POWER	PROVIDE POWER TO ENERGIZE AND CONTROL THE SPACECRAFT POWER SUBSYSTEM BUS IN SOLAR ARRAY SIMULATION MODE, OR IN AN UNCONDITIONED MODE WHERE LONG CABLE RUNS ARE UNVALUED (i.e., T/V).
	MONITOR PERTINENT POWER SUBSYSTEM PARAMETERS AND CONTROL SOLAR ARRAY SIMULATORS.
ORDNANCE	SIMULATE THE SPACECRAFT ORDNANCE LOAD (SQUIBS) AND MONITOR THE RESPONSE OF THE ORDNANCE CIRCUITS
	CONTINUOUSLY MONITOR AND DISPLAY THE STATUS OF ALL SPACECRAFT ORDNANCE CIRCUITS FOR VERIFICATION DURING TEST OPERATIONS
	PROVIDE MONITORING AND CONTROL FOR SIMULATION OF SPACECRAFT/ INTERSTAGE SEPARATION.
ATTITUDE CONTROL	PROVIDE SIGNAL TO SIMULATE THE OUTPUTS OF THE SPACECRAFT SUN SENSOR ASSEMBLY AND THE CONSCAN
	SIMULATE THRUSTER VALVE DRIVER LOAD AND MONITOR RESPONSE OF THRUSTER CIRCUITS
	SUN SENSOR STIMULATOR WILL BE PROVIDED AS A SELF-CONTAINED UNIT, BUT DRIVEN FROM THE SIMULATION SOURCE.
MISCELLANEOUS SIGNAL DISTRIBUTION	PROVIDE THE NECESSARY BUFFERING AMPLIFIERS AND TEST CONNECTORS FOR INTERFACING WITH SELECTED SPACECRAFT HARDLINES AND D DISTRIBUTING HARDLINES TO OTHER STS CONSOLES.
TIME	DISPLAY TIME OF DAY.

- Power monitor
- Digital volt meter (voltage)
- DVM (current)
- Ordnance control and monitor
- Console power supply
- Ordnance power supply
- Primary power control
- Time display
- Oscilloscope
- Solar array simulator
- Solar array simulator power supply
- Spacecraft external power supply.

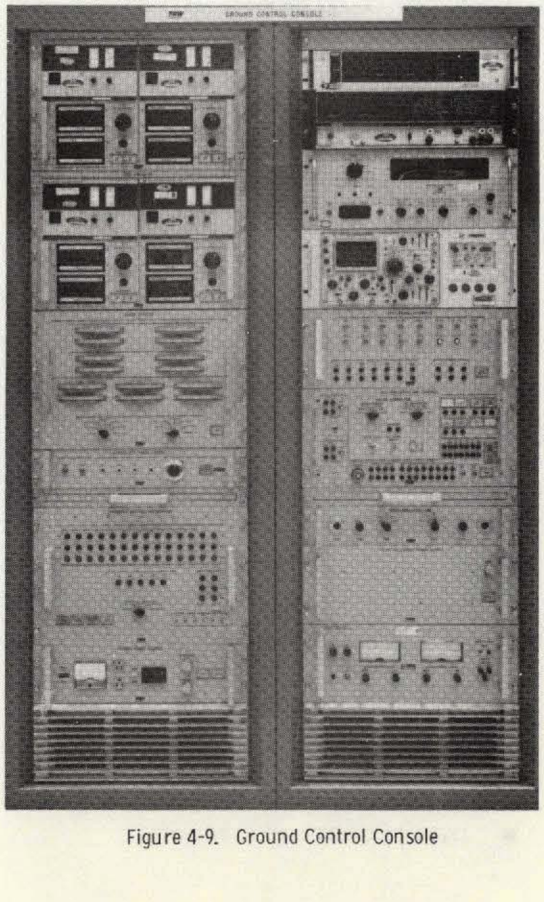


Figure 4-9. Ground Control Console

The following items are not shown in the figure because they will be external to the rack:

- Ordnance load simulator
- Thruster simulator
- Sun sensor stimulus.

Table 4-3 lists the modifications required to the above equipment.

4.3.1.4 Telemetry Data Console and Data Format Generator

The telemetry data console (TDC) will consolidate the control and monitoring of digital and discrete data that has been received from the spacecraft via the telemetry data transmission equipment. The functional requirements are:

- Demodulate downlink subcarrier into serial (PCM) data
- Condition data, establish clock, bit, word, frame, and format sync
- Load PCM data (in blocks of 24 parallel bits) into the ADPE direct input/output channel together with data rate clock

  Spacecraft status display

  Bit rate

  Format

  Mode

  Two digiswitch-controlled displays capable of selecting two 6-bit words in any telemetry format (nonsystematic convolutional)

  Generate or translate and display time code information.

- The data format generator (DFG) provides simulated telemetry data for self-test coded (nonsystematic convolutional coding) or uncoded, and with biphase PCM or NRZ-L data.

The TDC and DFG shown in Figure 4-10 contain the following items:

Telemetry Data Console

- Time code translater/generator

Table 4-3. Ground Control Console Equipment Modification

EQUIPMENT	MODIFICATION
THE TEST SIGNAL INTERFACE UNIT	WILL PROBABLY REQUIRE MODIFICATION TO BE COMPATIBLE WITH THE HARDLINE MONITOR POINTS SELECTED FOR TESTING ON THE ORBITER AND PMS.
ACS SIMULATION AND CONTROL	MODIFY TO SIMULATE PIONEER VENUS SUN SENSOR OUTPUT. PROVIDE TWO PULSES PER SUN CROSSING WITH MANUAL CONTROL OF TIME BETWEEN THE TWO PULSES TO SIMULATE DIFFERENT SUN ASPECT ANGLES. ELIMINATE THE STELLAR SENSOR SIMULATION AND STIMULUS CONTROL. ADD SIMULATION OF DMA INDEX AND RATE PULSES. INCREASE THE THRUSTER MONITOR CAPABILITY FROM SIX TO EIGHT THRUSTERS (EACH THRUSTER HAS REDUNDANT SOLENOIDS).
POWER MONITOR	ELIMINATE RTG MONITOR POINTS. ADD POINTS FOR MONITORING PROBE EXTERNAL VOLTAGE AND CURRENT AS SUPPLIED BY THE BUS. ELIMINATE TWTA POWER MONITOR POINTS. ADD POINTS FOR MONITORING THE SPACECRAFT BUS VOLTAGE AND CURRENT AND BATTERY TEMPERATURE.
ORDNANCE CONTROL AND MONITOR	THE EXISTING UNIT THAT HAS THE CAPABILITY TO CONTROL AND MONITOR 14 SQUIB CIRCUITS) WILL BE REPLACED WITH A LARGER ONE WITH THE CAPABILITY TO CONTROL AND MONITOR UP TO 28 SQUIB CIRCUITS FOR THE PROBE BUS.
ORDNANCE POWER SUPPLY	WILL BE REPLACED WITH A HIGHER CURRENT SUPPLY TO BE COMPATIBLE WITH THE INCREASED MONITORING REQUIREMENTS.
SOLAR ARRAY SIMULATOR	THIS WILL BE A TRW CAPITAL UNIT TO REPLACE THE RTG SIMULATORS. THE SAS SHOULD SIMULATE THE KNEE OF ARRAY TRANSITION FROM CONSTANT CURRENT TO CONSTANT VOLTAGE. THE SAS SHOULD GENERATE A MAXIMUM OPEN CIRCUIT VOLTAGE OF 55 VOLTS AND A MAXIMUM SHORT CIRCUIT CURRENT OF 12 AMPS. THE SAS OPEN CIRCUIT VOLTAGE AND SHORT CIRCUIT CURRENT SHOULD BE ADJUSTABLE TO SIMULATE THE ARRAY CHARACTERISTICS AT THE NEAR-EARTH AND NEAR-VENUS LOCATIONS.
SOLAR ARRAY SIMULATOR POWER SUPPLY	THIS WILL BE A TRW CAPITAL UNIT TO PROVIDE THE DC POWER TO THE SOLAR ARRAY SIMULATOR WHERE IT IS CONDITIONED FOR SIMULATING THE SPACECRAFT SOLAR ARRAY.
SPACECRAFT EXTERNAL POWER SUPPLY	THIS WILL BE A NEW ITEM USED TO SUPPLY POWER TO THE BUS IN A VOLTAGE-REGULATED MODE SUCH THAT THE BUS VOLTAGE MAY BE ADJUSTED OVER SPECIFIED LIMITS. IT WILL ALSO FEATURE CURRENT LIMITING FOR PROTECTION OF THE SPACECRAFT CIRCUITS AS WELL AS SELF-PROTECTION OF THE POWER SUPPLY.
SPACECRAFT POWER CONTROL	THIS NEW UNIT WILL CONTROL THE SELECTION OF EITHER THE SOLAR ARRAY SIMULATION MODE OR THE UNCONDITIONED POWER MODE. IT WILL ALSO CONTAIN CIRCUITRY TO ENSURE A SLOW RAMP FUNCTION POWER TURN-ON OF GREATER THAN 100 MILLISECONDS.
ORDNANCE LOAD SIMULATOR	THIS ITEM WILL BE MODIFIED TO INCREASE THE QUANTITY OF SIMULATED SQUIBS FROM 14 TO 28 FOR THE PROBE BUS.
THRUSTER SIMULATOR	THIS ITEM WILL BE MODIFIED TO INCREASE THE QUANTITY OF SIMULATED THRUSTERS FROM SIX TO EIGHT. THE MODIFICATION WILL CONSIST OF CHANGING THE CABLING THAT CONNECTS THE THRUSTER SIMULATORS TO THE SPACECRAFT IN PLACE OF THE ACTUAL THRUSTERS. FOUR OF THE PIONEER 10 AND 11 DUAL THRUSTER SIMULATOR UNITS WILL BE REQUIRED FOR PIONEER VENUS.
SUN SENSOR STIMULUS	THIS ITEM WILL BE MODIFIED TO BE COMPATIBLE WITH THE INTELSAT III SUN SENSOR.

- Fixed word display
- PSK demodulator EMR 2726
- Decoder/buffer
- PCM bit sync EMR 2720
- Spacecraft status display
- Console power control
- PCM decommutator EMR 2746
- Decimal display EMR 2756
- Power supply +28 VDC
- Power supply -15 VDC
- Power supply ±15 volts
- Intercom
- Monitor assembly, power supply.

Data Format Generator

- Minicomputer
- PSK data modulator
- ASR-33
- Primary power control.

Modifications to these items will be as follows:

- Fixed word display. The fixed word display will strip out and display the fixed words in the telemetry data. These words are: bit rate, mode, and format identification (ID), and are always located in the same position in the telemetry data stream. This item will be modified to be compatible with the Pioneer Venus telemetry format.

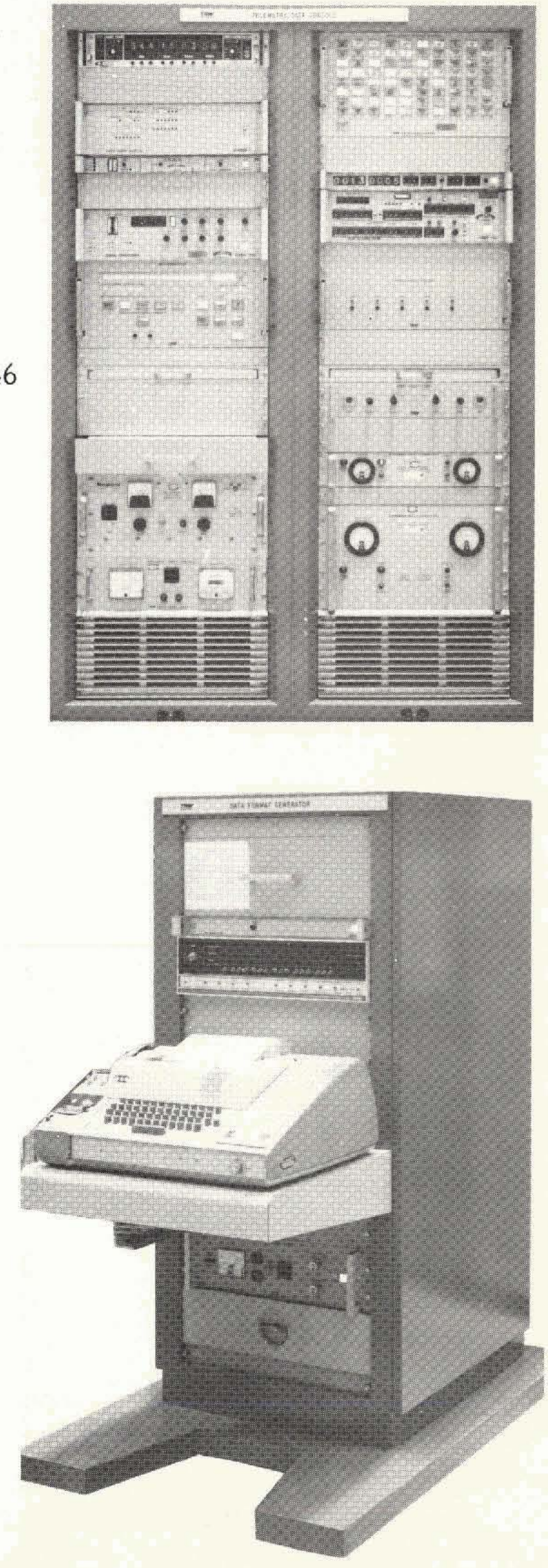


Figure 4-10. Telemetry Data Console and Data Format Generator

- Spacecraft status display. The spacecraft status display receives spacecraft status data in parallel words from the ADPE. These words are decoded and displayed on indicators, each identifying a discrete bit of information regarding the spacecraft's operation. This item will require modification to display the desired Pioneer Venus spacecraft status data.

- Data format generator. Software will be modified to accommodate changes in the telemetry format.

4.3.1.5 Test Conductor's Console

The test conductor's console (TCC) will consolidate the control and monitoring of critical spacecraft functions to a single convenient location. It will permit the test conductor to control commands sent to the spacecraft, to review selected telemetry data, and to communicate with the rest of the test crew during major system tests such as integrated system tests (IST), thermal-vacuum, countdown, etc. The TCC performs these functions in conjunction with the RF console, the automatic data processing equipment (ADPE), the telemetry data console (TDC), and the ground control console (GCC). The functional requirements are:

- CRT display for displaying selected spacecraft telemetry data including fixed displays of bit rate, format, and mode
- Command encoder to enable generating command codes to modulate the uplink transmitter housed in the RF console
- Monitors selected STS input/output signals
- Two digiswitch-controlled telemetry displays capable of selecting any two 6-bit words in any telemetry format
- House the intercommunications system
- Display time of day.

The TCC shown in Figure 4-11 contains the following items:

- Oscilloscope
- Signal monitor
- Intercommunication
- Remote time display
- TV display select
- TV display

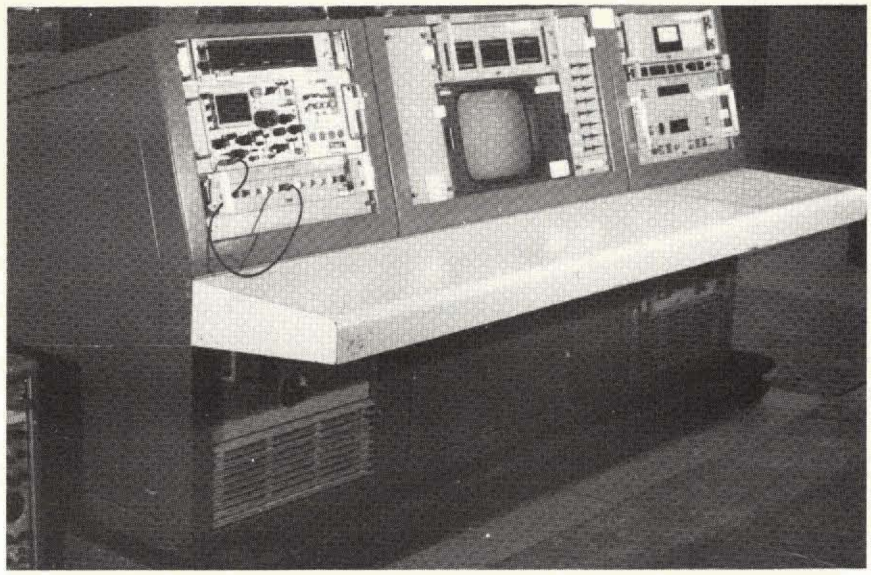


Figure 4-11. Test Conductor's Console

- Decimal display EMR 2756
- Command encoder
- Receiver signal strength
- Primary power control.

No modifications are required for this equipment.

4.3.1.6 EGSE Peripheral Equipment

The peripheral equipment performs spacecraft tests as well as tests the STS. The functional requirements are:

- Provide breakout test points in the spacecraft cabling
- Provide fuses to protect the spacecraft units and cabling during initial integration; including capability to short out the fuses
- Measure the RF leakage from the spacecraft bus and probes communication subsystems
- Measure the STS hardline S-band command carrier power level at the point of interface with the spacecraft
- Interconnect the STS consoles
- Connect the STS to the spacecraft
- Simulate spacecraft bus and probe power subsystem signals during validation of the STS power monitor

- Simulate spacecraft PCU during validation of the STS solar array simulator

- Simulate spacecraft power bus load during validation of the STS spacecraft external power supply.

In addition, as part of the peripheral, a blockhouse unit will be used at the launch site during the countdown when standby power must be supplied through the launch vehicle umbilical cable. It will control and monitor the SRM and ordnance safe/arm device on the orbiter and PMS. Functional requirements are:

- Provide standby power to spacecraft via the launch vehicle umbilical

- Monitor voltage and current supplied to spacecraft

- Provide SRM ignitor safe indicator (orbiter mission)

- Provide SRM ignitor arm indicator (orbiter mission)

- Provide voice communications to other launch site locations.

The EGSE peripheral equipment includes the following items:

- Inline series fuse box

- STS power validator

- STS signal validator

- Cable set

- Spacecraft ordnance checkout unit

- Blockhouse unit.

Modifications to this equipment will be as follows:

- STS power validator. This will be a new item which will be used in conjunction with external, general-purpose capital test equipment to validate the power monitor, solar array simulator, spacecraft external power supply, and in-flight jumper simulator of the STS. It will simulate the spacecraft bus and probe power subsystems signals that are monitored by the power monitor, including power bus voltages and currents as well as battery temperatures. The STS power validator will include test points for checking the operation of the power monitor control circuits. Adjustable resistive loads will be included to validate the solar array simulator and spacecraft external power supply.

- STS signal validator. This item will be modified to permit validation of the modified ACS simulation and control drawer and the new ordnance control and monitor drawer. Modifications will include elimination of signals associated with the stellar reference assembly and RTG's. Circuits will be added to enable validation of the simulated DMA index and rate pulses, the increased thruster monitor capability, and the increased ordnance monitor capability.

- Cable set. The cables that interconnect the STS consoles will be modified to be compatible with the STS and spacecraft changes. New cables will be required to interconnect the STS to the spacecraft.

The blockhouse unit will be primarily TRW capital equipment except for the intercom and the SRM safe/arm monitor and control. The intercom will be supplied GFE by ETR. The SRM, ordnance, safe/arm monitor, and control will be a new item developed for the Pioneer Venus program.

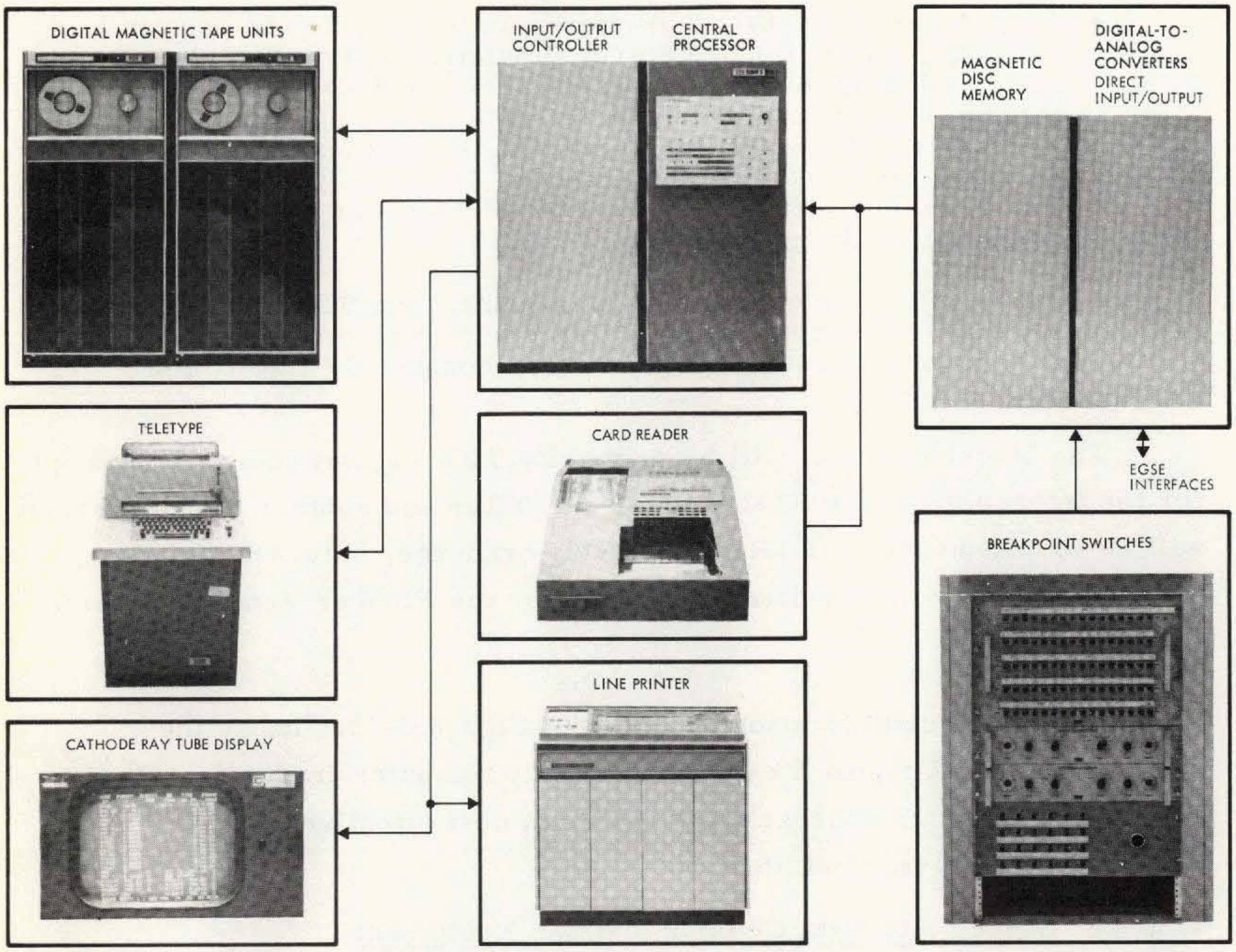
4.3.1.7 ADPE

The ADPE configuration is shown in Figure 4-12. It has the Pioneer 10 and 11 Sigma 5 configuration. A computer tradeoff study determined that this approach was the most cost effective. Section 4.4 details the ADPE tradeoff study.

4.3.1.8 Launch Site Data Communications Equipment

During launch operations, several consoles of the STS will remain in the hangar while the spacecraft is on the launch pad. This will require the use of some launch site communications equipment to provide reliable data transmission between the spacecraft and the consoles in the hangar in both the RF radiated and hardline modes of operation as was done on Pioneers 10 and 11. This equipment will be provided as GFE by ETR and consists of the following:

- S-band antennas at launch pad
- S-band high-gain antenna at the hangar
- Low loss coax cables for S-band
- Video line driver at the hangar with 10-channel wideband VCO/demodulator
- Video line driver at launch pad
- Low loss video cables between hangar and launch pad.



AUTOMATIC DATA PROCESSING EQUIPMENT (ADPE)

GENERAL PRUPOSE COMPUTER
CONTROLLED FROM:
- CARD READER
- TELETYPE
- DISC FILE
- MAGNETIC TAPE
- BREAKPOINT SWITCHES

TO PERFORM:
- TELEMETRY SYNCH AND DECODING
- DETERMINATION OF FORMAT/BIT RATE/MODE
- EXTRACT AND DISPLAY/PRINTOUT

ENGINEERING AND SCIENTIFIC DATA
- PRINTOUT AND AUDIBLE NOTIFICATION OF OUT OF LIMITS DATA
- AUTOMATIC CONTROL AND PRINTOUT OF COMMANDS
- DISPLAY/PRINTOUT TIME OF DATA ENTRY

DISPLAYS:
- LINE PRINTER
- CATHODE RAY TUBE DISPLAY

Figure 4-12. ADPE

The antennas will provide the RF link between the hangar and the launch pad. A coax cable will connect the hangar high-gain antenna to the RF console in the hangar. A second coax cable will connect the launch pad high-gain antenna to the medium-gain antenna.

The video line drivers and video cables will enable the hardline transmission of commands and telemetry between the hangar and the launch pad.

4.3.2 Probe EGSE

The study requirements for the probes EGSE were to analyze ground support operations, identify the major EGSE components, and establish a cost-effective design concept. Specifically, the EGSE is required to support probe subsystem level tests, systems integration and checkout, systems environmental tests, systems acceptance test, and spacecraft integration and prelaunch tests.

4.3.2.1 Design Concept

The basic design approach was to establish support equipment at the subsystem level. A test set for independent support of probe subsystems forms the basis of providing a probe system test set.

The subsystem test equipment provides a checkout capability to the component (black box) interface level. This provides simulation and response measurement functions for power, command, and data signals (with attendant fault isolation at the component level.

Upon integration of the flight subsystem hardware into a probe system, tests are performed using a bus/probe interface simulator (BPIS) to operate and control the probe. Necessary subsystem test equipment augments the BPIS to provide power, pyrotechnic simulation and monitor functions, battery charging, and RF/hardline data receiving, demodulation, and formatting for data processing.

An essential part of a cost-effective design approach was to use common equipment and design for the large and small probes. Also, the sharing of common EGSE between the probes and bus was a design objective to minimize the delivered EGSE. The basic design concept is shown in Figure 4-13.

The science instruments and associated EGSE are government-furnished equipment for the program and are not shown. This GFE will be used for science integration tests and for science calibration/stimulus during probe systems tests, with the exception of EGSE to test the mass spectrometer pyrotechnics.

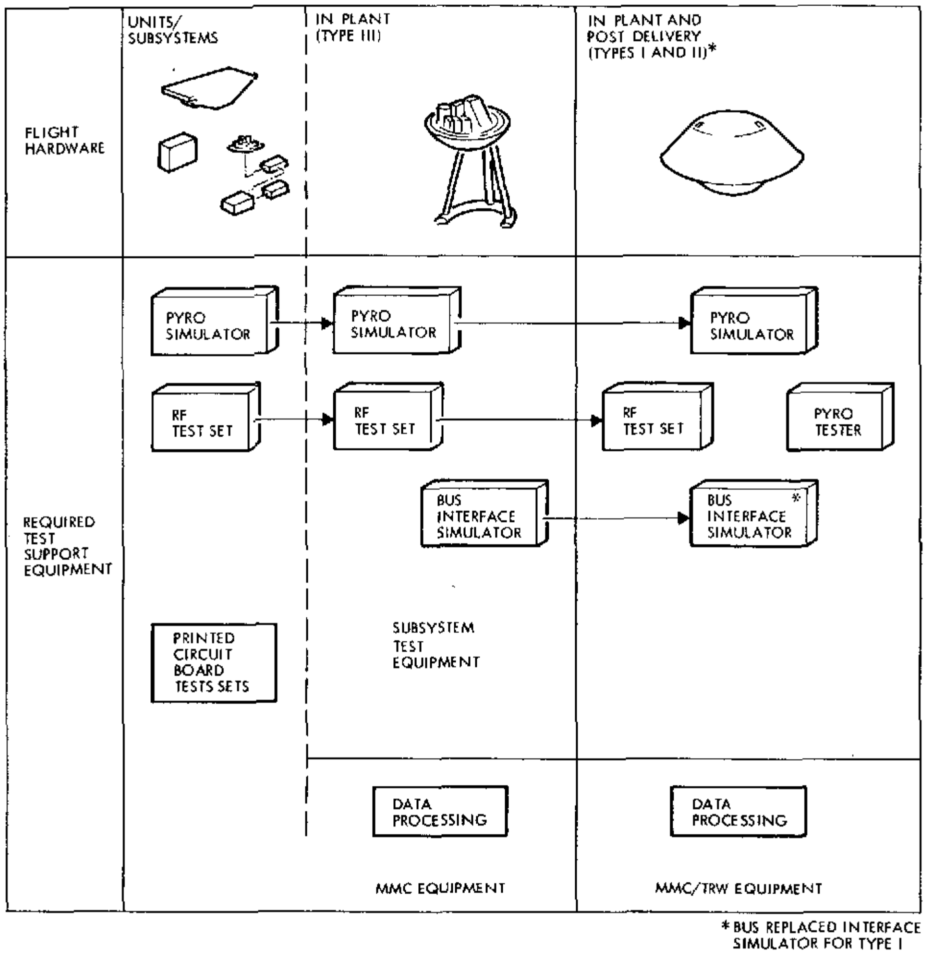


Figure 4-13. Probes EGSE Design Concept

4.3.2.2 Design/Cost Tradeoffs

Two areas were evaluated as design/cost tradeoffs, a manual versus automated EGSE system, and the variance in design for the Thor/Delta versus Atlas/Centaur probe configurations. The use of existing capital equipment or GFE was also considered.

Initially, an automated system was proposed using a general-purpose computer for both test sequencing and control, and data processing. Cost comparisons indicated a reduction in implementing the design with a manual control capability and computer data processing using capital equipment. The savings were primarily in deleting the computer and instrumentation test set design and hardware, and software design costs for computer test sequence control. Since a computer data processing

technique is used in both the TRW Pioneer 10 and 11 automatic data processing equipment and at the JPL Deep Space Network (DSN), this method was chosen for the probes EGSE. This provides continuity in the processing of probe data from in-plant test, through spacecraft integration test, and the mission operations phases. Other considerations also favor the automatic (computer) data processing method; i.e., the probe data system consists of several formats that are automatically sequenced during tests; decoding of the convolutional coded data is readily performed by the computer; use of existing data processing facilities in-plant eliminates hardware decoder/decommutator costs; and software design for probe data processing can be developed early for subsequent use at TRW/DSN.

The design was evaluated for both the Thor/Delta and Atlas/Centaur. Some additional test point access may be available in the Atlas/Centaur configuration at the subsystem test level; but the conclusion was that the launch vehicle has no significant impact on the EGSE since the bus/probe interface and probe subsystem functions, which affect design, are essentially the same in each case.

The optimal design concept requires a minimum of equipment design, uses existing capital equipment for in-plant test support and data processing, and uses the existing TRW ADPE for data processing subsequent to probe delivery. This design employs the desirable features of both manual test equipment operation and control, and of automatic data processing techniques.

4.3.2.3 EGSE Functional Items

The probes EGSE functional items are listed in Table 4-4. The functional schematic is shown in Figure 4-14. The table shows the equipment usage, equipment source, and required quantities of each item. These items provide the general test functions as noted below.

RF Test Set

The function of the RF test set is to receive, demodulate, and condition the probe digital data stream via the S-band carrier or hardline link. The demodulated signal is bit synchronized and a serial-to-parallel buffer/register provides the interface for the computer data processing equipment. An analog-to-digital converter is included in the bit synchronizer

Table 4-4. Probes EGSE Functional Items

EQUIPMENT LIST \ SOURCE USAGE DATA	UTILIZATION							USED AT					DESIGN STATUS			SOURCE	
	IN-PROCESS TEST	DESIGN TEST	INTEGRATION SUPPORT	INTEGRATION TEST	QUALIFICATION TEST	ACCEPTANCE TEST	COMPATIBILITY TEST	MMC-DENVER	MMC-ORLANDO	TRW	LAUNCH SITE	QUANTITY	EXISTING	MODIFIED	NEW	CAPITAL EQUIPMENT	CONTRACT
1. RF TEST SET	X	X	X	X	X	X	X	X	X	X	X	1			X		X
2. BUS/PROBE INTERFACE SIMULATOR*				X	X	X		X	X	X	X	1			X		X
3. PYRO SIMULATOR*				X	X	X	X	X		X	X	1			X		X
4. PYRO TESTER*					X	X	X	X		X	X	1	X			X	
5. DATA PROCESSING STATION**				X	X	X	X	X	X	X	X	1	X			X	
STATUS SUMMARY - PERCENTAGE													50	10	40	60	40

*DELIVERED WITH PROBES
**USE TRW PIONEERS 10 AND 11 ADPE SUBSEQUENT TO PROBE DELIVERY

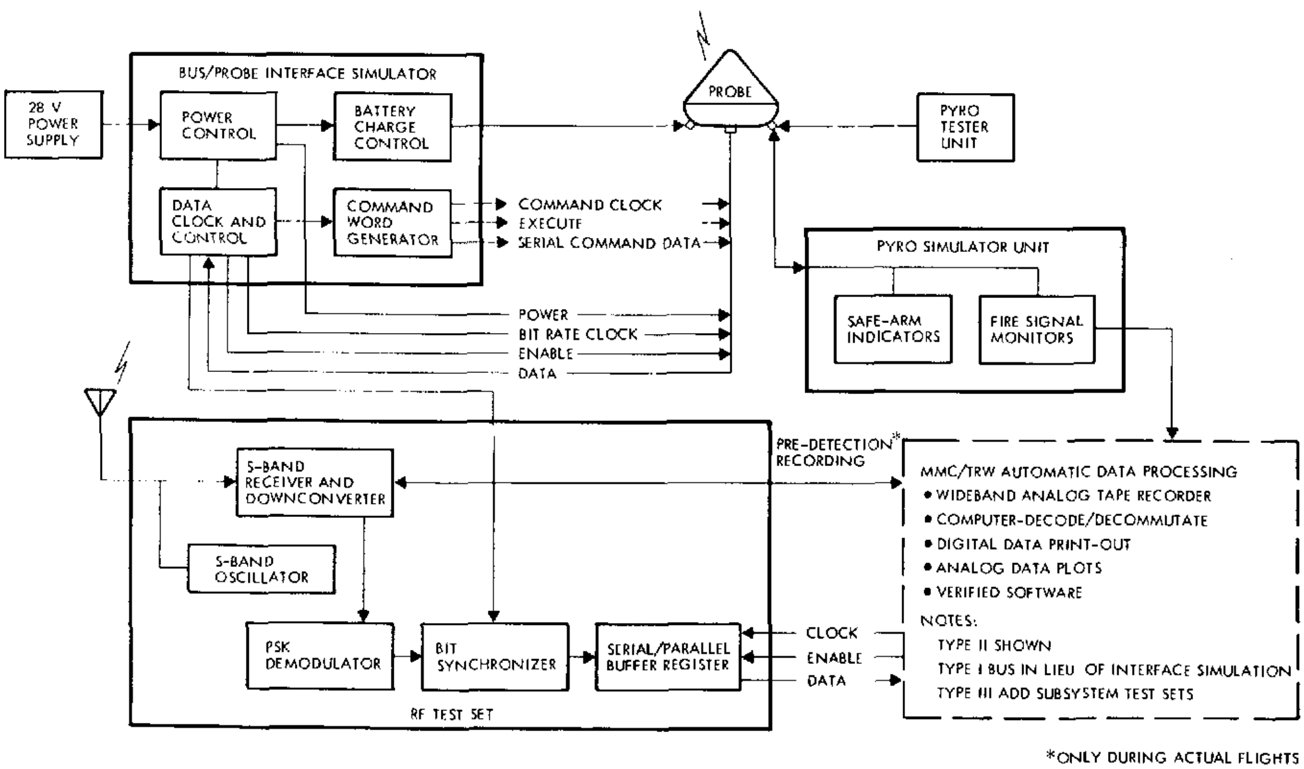


Figure 4-14. EGSE Functional Schematic

to provide signal quantitizing for soft decision (low signal-to-noise) decoding. This test set will support all test types (I, II, and III).

Bus/Probe Interface Simulator (BPIS)

The BPIS provides all bus power, command, and data interface functions during probe systems tests. This unit has the necessary equipment and circuitry to provide the power, command, and data functions shown in Figure 4-14. The BPIS is used for all assembled probe stand-along Type II tests.

Pyrotechnic Simulator

The pyro simulator provides external simulation of pyro devices in the probe. In addition to simulation of the pyro devices, this unit monitors safe/arm commands and the condition of the circuitry, and also provides a measurement source for firing signals to an external recorder. This unit is used in Type I and II tests.

Pyrotechnic Tester

The pyro tester verifies the condition of live pyrotechnic initiators. This unit is used for Type I and II support.

Data Processing Station

The data processing station provides decoding, decommutation, quantitizing, and display of probe system data via the RF test set. The basic equipment is in existing laboratories and consists of a general-purpose computer and peripherals, a wideband analog tape recorder, D/A converters, and direct write oscillographs. Martin Marietta capital equipment will be used for all in-plant Type II and III test support; the TRW Pioneer ADPE will be used for data processing subsequent to probe delivery, Type I and II tests.

General Electronic Test Equipment

In addition to the items described above, general-purpose electronic test equipment from existing capital equipment inventory will be used to support Type II and III in-plant probe tests.

4.3.2.4 Schedule and Utilization

The utilization of the probes EGSE is shown in Table 4-4. The schedule is shown in Figure 4-4 (Section 4.2). In developing the probe

program plan, the quantities of EGSE are an important factor in the decision process of establishing a low-cost program. The quantities of an individual end item of equipment along with the necessary test crew(s) became a function of sequential or concurrent fabrication and test of the large and small probes. The indicated schedule minimizes costs since it is accommodated by one set of EGSE.

4.3.2.5 Software Requirements

The basic probe serial digital data stream is processed at Martin Marietta, TRW, and the launch site. Therefore, variations in the computer software, which performs the data decoding-decommutation, must be provided.

Generally, two different computers may be used for probe data processing at the test sites stated above. In addition, the probe data will be interleaved with bus data in the spacecraft configuration tests at TRW and CKAFS. Therefore, some five data processing routines for the probe system may be involved.

To minimize software design and provide a universal data processing base for the probes, the decode-decommutation tables will be written in Fortran language.

Martin Marietta will adapt these tables for the computer system used for data processing prior to probe delivery.

These data will be supplied to TRW for adaption to the computers used for probe test, post-delivery, and for spacecraft testing at TRW and CKAFS.

Since the probe software interfaces with bus software, the probe software design will use existing TRW Pioneer routines where possible. Software design integration will be effected between Martin Marietta/ TRW to assure that common design efforts are not duplicated, and that software requirements are provided efficiently.

4.3.3 Spacecraft Interface Simulator

The Pioneer spacecraft simulator provides the means for exercising all of the scientific experiments and printing out the resultant data. To assure compatibility between the experiments and the spacecraft, the

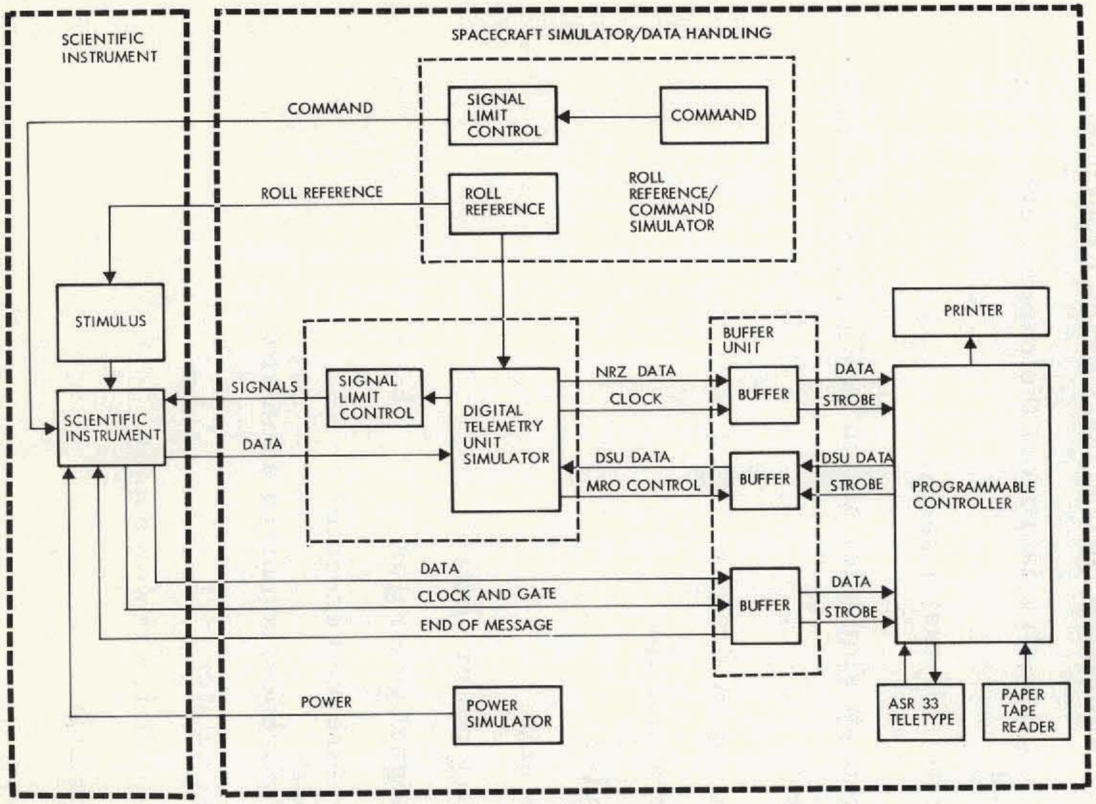
simulator duplicates the spacecraft/experiment interface. It provides power, timing signals, bilevel signals, roll rate signals and command signals. It accepts digital, analog, and bilevel data and translates the data into numerical values and formats for printout.

The spacecraft simulator has capability for self-test and software checkout. The functional requirements are:

- Provide power to the experiments
- Provide experiment power on/off commands
- Provide discrete commands to the experiments
- Provide roll reference pulse to experiments
- Provide simulated DTU signals to the experiment
- Receive telemetry data from the experiment, format, and printout
- Generate self-test signals.

The spacecraft simulator shown in Figure 4-15 contains the following items:

- Power supply drawer
- Power simulator
- Computer
- Buffer unit
- Primary power control
- Test signal generator
- Roll reference generator
- Roll reference/command simulator
- DTU simulator
- DTU simulator power supply
- Line printer
- Teletype



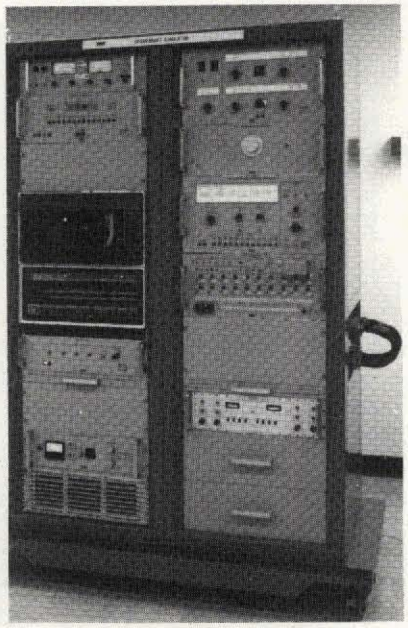


Figure 4-15. Spacecraft Interface Simulator

- Paper tape reader
- Cable set.

The spacecraft simulator will require the following modifications:

- Power supply. Modify to provide proper output power for the Pioneer Venus experiment
- Cable set. New cables will be required to simulate the Pioneer Venus spacecraft cables
- Power simulator. Modify to provide the proper output impedance
- DTU simulator. Modify to allow changes of the telemetry format and bit rate
- Buffer unit. Modify to provide for DSU simulation changes
- Software. Modify for telemetry format changes.

4.4 ADPE TRADEOFF STUDY

This section delineates the results of a study to determine an ADPE configuration which is most cost effective and best meets the Pioneer Venus test requirements. Two general approaches of hardware implementation with their associated software were examined and compared. The first is the centralized processor approach (separate midicomputer, i.e., Sigma 5), which was used on Pioneers 10 and 11 and is representative of most current ADPE configurations for spacecraft testing. The second is the distributed processor approach, which utilizes recent technological achievements in minicomputers.

4.4.1 ADPE Test Requirements

4.4.1.1 General System Requirements

The ADPE must be capable of supporting the engineering and scientific instrument data processing requirements for both the orbiter and probe mission spacecraft. This tradeoff study assumes the orbiter mission and probe mission systems test programs will be performed serially. The ADPE will not be required to perform in parallel the real time data readout processing functions on all scientific instruments, or either the orbiter, the probe bus, or the individual probes. The capabilities for

parallel real time data readout (line printer speed limitations) will be established after the receipt of instrument programming and system test requirements.

The ADPE must also be able to process probe telemetry engineering and instrument data when the probes are undergoing unit level test at TRW or at the launch site.

4.4.1.2 System Functional Requirements

The ADPE must, as part of the STS (Figure 4-16), continuously monitor telemetry data, and functionally provide the following capabilities:

- Verify telemetry word and frame synchronization
- Determine the format of the telemetry

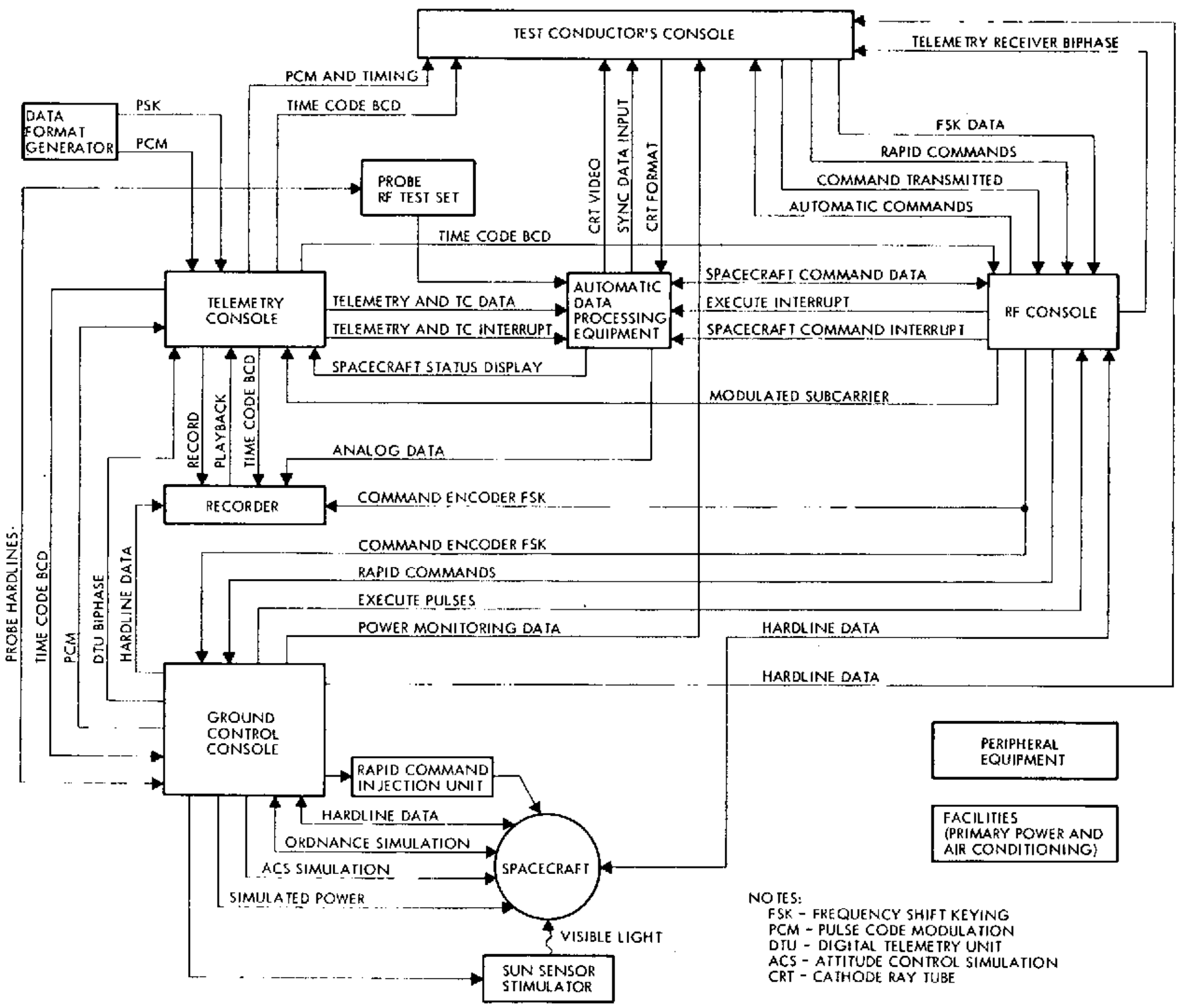


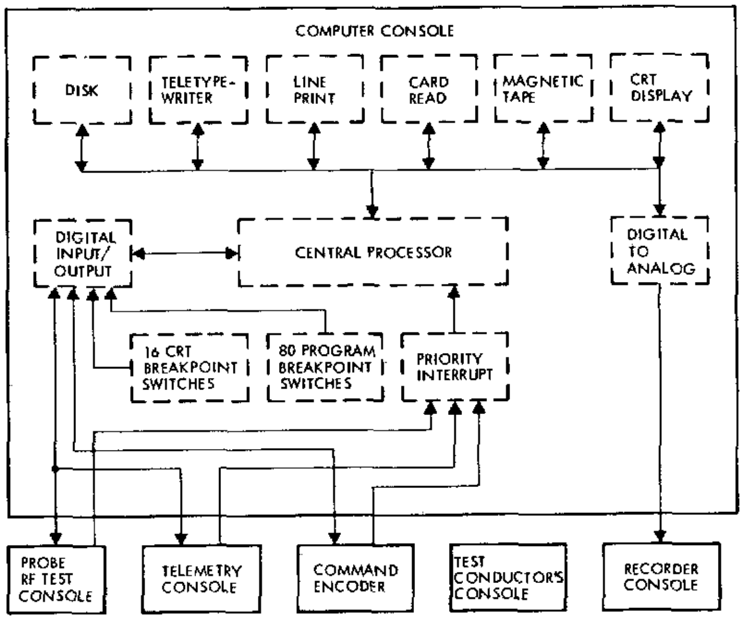
Figure 4-16. EGSE System Test Set Block Diagram

- Determine the bit rate of the telemetry
- Determine if the telemetry is coded
- Strip and derive decoded from coded telemetry data
- Assess the quality of coded telemetry data being processed
- Determine the operating mode of the spacecraft
- Determine the spacecraft configuration
- Extract engineering and scientific instrument words
- Convert binary data words to binary or binary coded decimal and octal formats suitable for display and recording
- Determine the value of limit-checked words and provide notification of out-of-limit conditions
- Evaluate time data entry and provide notification of improper entry
- Process and print out data received from the card stack reader
- Process and perform instructions from the input/output typewriter and the sense switches
- Print out engineering and scientific instruments data in suitable decimal and octal equivalents
- Provide alarm indication of abnormal conditions
- Automatically transmit commands by computer control
- Record historical data on magnetic tape.

4.4.2 Technical Requirements

4.4.2.1 Hardware Requirements

A basic definition of the minimal ADPE hardware requirements can be established, although the implementation may vary between various proposed configurations. An overall block diagram of the ADPE components is shown in Figure 4-17. The hardware requirements for the ADPE may be divided into three general categories, central processor, peripheral, and EGSE interfaces.



CRT DISPLAY AND CRT BREAKPOINT SWITCHES LOCATED AT THE TEST COND CONSOLE

Figure 4-17. ADPE Interface Block Diagram

Central Processing Unit (CPU)

The CPU may be considered one main central processor (i.e., Sigma 5), or, in the case of a distributed minicomputer system, as a group of processors. The system CPU should minimally contain, in addition to its full complement of operating instructions, the following characteristics:

- Core memory capacity: 128 K bytes (8 bits/byte) expandable to 256 K bytes
- Memory cycle time: 1.0 microsecond or less
- Hardware multiply – divide
- Priority interrupt system with an interrupt acknowledge of less than 50 microseconds
- At least one direct memory access (DMA) input/output channel
- Real time clock
- Console panel for register display, register load, step execution, and power on/off control.

Peripherals

The following peripherals are required to meet the functional requirements.

Typewriter. A typewriter is required for communication between the operating personnel and the computer. The typewriter must be capable of entering data into the CPU and of typing data received from the CPU at a rate of ten or more characters per second.

Magnetic Tape Recorder. Two magnetic tape units are required. The tape units must be 9-track industry compatible units on which the data is read and written at 800 bits/inch. A full complement of read orders, space order, rewind order, erase orders, and sense orders must be available in the tape unit.

The magnetic tape units will be used for permanent storage of all programs in case of a catastrophic failure that alters disc or core memory. For software development, all source and object programs will be stored on tapes. The tape units will record telemetry test data for permanent file and off-line processing.

Line Printer. There must be at least one high speed line printer, capable of printing 132 character lines at 1000 lines/minute or greater.

The printer must perform the following tasks:

- Print out processed (formatted and converted into meaningful units) spacecraft subsystem data derived from telemetry inputs
- Print out processed experiment data
- Periodically print out all commands sent to the spacecraft in the order sent
- Print out program assembly listings
- Off-line print out test procedures on reproducible masters
- Print out memory dumps and other diagnostic printouts.

Card Reader. A card reader is required to read standard IBM compatible prepunched cards. The card reader speed shall be at least 200 cards per minute. The card reader may be used for software development, test parameter modifications, calibration factor specification, test mode identification, and input of automatic sequential commands.

CRT Alphanumeric Display. At least one alphanumeric display is required, with provisions for at least two additional slave monitors. The CRT should make available 20 lines of 40 characters. The CRT is

required to display computer-processed (formatted and converted into meaningful units) spacecraft subsystem data derived from telemetry. It can display processed experiment data, input command sequence data, critical bus voltage via telemetry, and abnormal conditions.

Disc Memory. A disc memory with a memory capacity of at least 0.7 mega bytes (8 bits/byte) is required. The average data access time should be less than 40 milliseconds. The disc will store operational programs, offline programs, and assembly programs off line. All operational programs must be stored on the disc in such a way that they can be quickly read into core memory.

EGSE Interfaces

The ADPE is required to interface with the existing Pioneers 10 and 11 EGSE. It is desirable to keep any EGSE modifications to a minimum to accomplish this task. The EGSE interfaces may be divided into digital input/output, priority interrupts, and analog.

Digital Input/Output. The signal levels of the EGSE digital input/output is essentially TTL compatible, with drive capabilities for 15.24 meters (50 feet). The data format presented to the Sigma 5 is shown in Figure 4-18. The following table identifies the digital input/output quantity requirements.

Signal Name	Source/Destination	No. of Bits
Digital inputs		
Telemetry data	Telemetry data console	32
Time code data	Telemetry data console	32
Command modification	Command encoder	16
Breakpoint switches	Breakpoint switch unit	80
CRT display switches	CRT switch panel (TCC)	16
Probe telemetry data	Probe test set	32
Digital outputs		
Spacecraft status display/sync mode	Telemetry data console	16
Command encoder (Auto mode)	Command encoder	16
Probe command	Probe test set	16

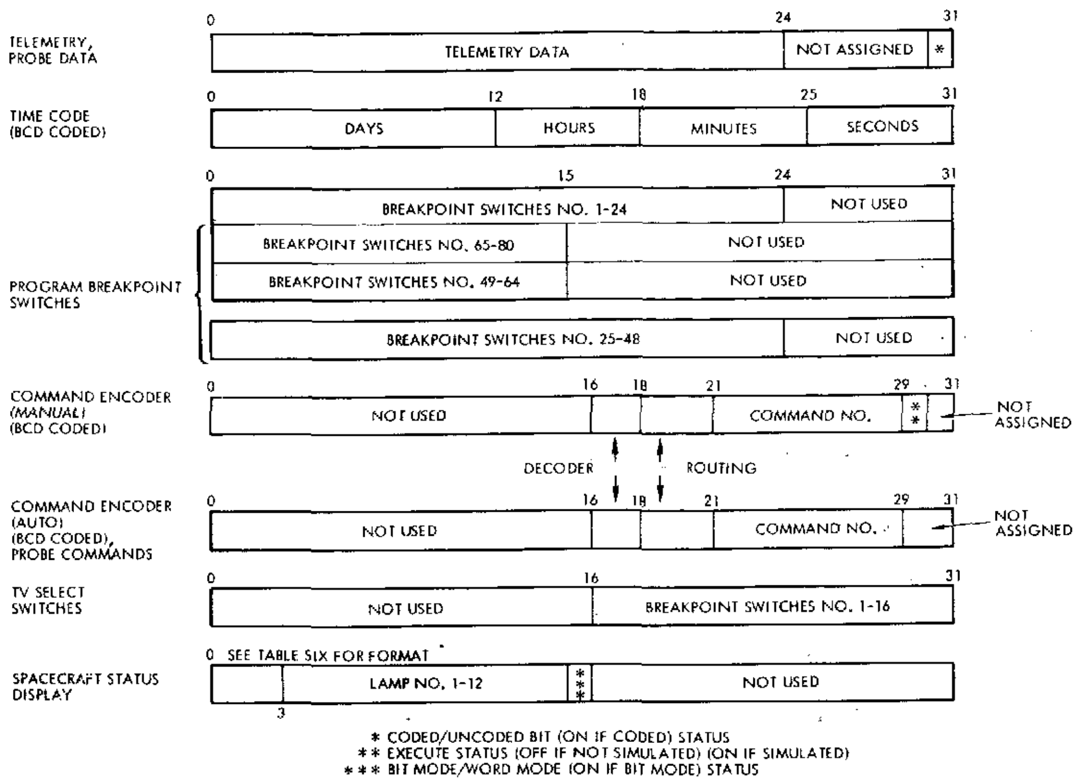


Figure 4-18. Contents of Sigma Input/Output Register for each Device using DIO

Priority Interrupts. The priority interrupt signals are nominally 4.0-microsecond, positive-going, TTL-compatible pulses. The nine required external interrupt signals are identified in the following signal assignment table:

Signal Name	Source
Manual end of message	Command encoder
Automatic end of message	Command encoder
Execute pulse	Command encoder
Telemetry data transfer pulse	Telemetry data console
Frame identification pulse	Telemetry data console
Subframe identification pulse	Telemetry data console
Time code generator	Telemetry data console
Probe data ready	Probe test set
Probe command ready	Probe test set

Digital-to-Analog (D/A) Converters. The ADPE is required to provide 16 D/A channels to permit telemetry data to be recorded on strip-charts. The D/A converters should have at least 8-bit resolution.

4.4.2.2 Software Requirements

In addition to the TRW-generated operational software, a set of computer vendor deliverable programs will minimally contain the following programs:

- Operating systems
- System generation routines
- Utilities

    *Assembler/concordance

    Linkage editor

    Conversion routines

    Input/output subroutines

    System diagnostics

    *Machine diagnostics

    *Debug aids

    *Any necessary loaders/dump routines

    *Media conversion routines (card/print, card/magnetic tape, etc.)

    Trace routines

    *Source editor

    Fortran.

4.4.3 Candidate Machine Evaluation

This section presents a system description of both the centralized processor approach and the distributed processing approach. For each approach, hardware and cost data have been accumulated from a representative computer manufacturer. No attempt has been made to survey the many computer manufacturers since the two selected manufacturers are typical of the industry.

Those marked * should be available in system and "stand-alone" versions.

4.4.3.1 Centralized Processor

The centralized processor approach utilizes one medium-sized computer to accomplish spacecraft testing. Processing is controlled by a single executive operating system that serializes operations according to priority.

The Pioneers 10 and 11 ADPE, which is representative of the centralized processor approach and meets Pioneer Venus requirements, is shown in Figure 4-19. It is characterized by a medium-sized CPU (Sigma 5), direct input/output (DIO) to EGSE, priority interrupts to EGSE, and an input/output processor (IOP) with the associated peripheral.

A list of the current XDS Sigma 5 equipment necessary to implement this configuration and associated prices are shown in Table 4-5.

4.4.3.2 Distributed Processing Approach

The distributed processing approach is characterized by the utilization of a multi-minicomputer configuration. These minicomputers are dedicated to specific tasks.

A distributed processing system configuration applicable to the Pioneer Venus ADPE is illustrated in Figure 4-19. It is comprised of: three independent process controllers, core memory of 64 K words (16 bits/word), and required peripheral and EGSE interfaces. The three processors are functionally divided as follows:

- The system controller primarily is devoted to those tasks related to interfacing with the test operator, input/output control, and system control analogous to the operating system of the centralized processor. Functionally, the system controller performs the following tasks:

  Operator interface

  Peripheral input/output control

  System priority

  Conversion of telemetry to formats suitable for printouts and displays

  Spacecraft command sequence control

  Historical data processing and recording

  Data time tagging facilities.

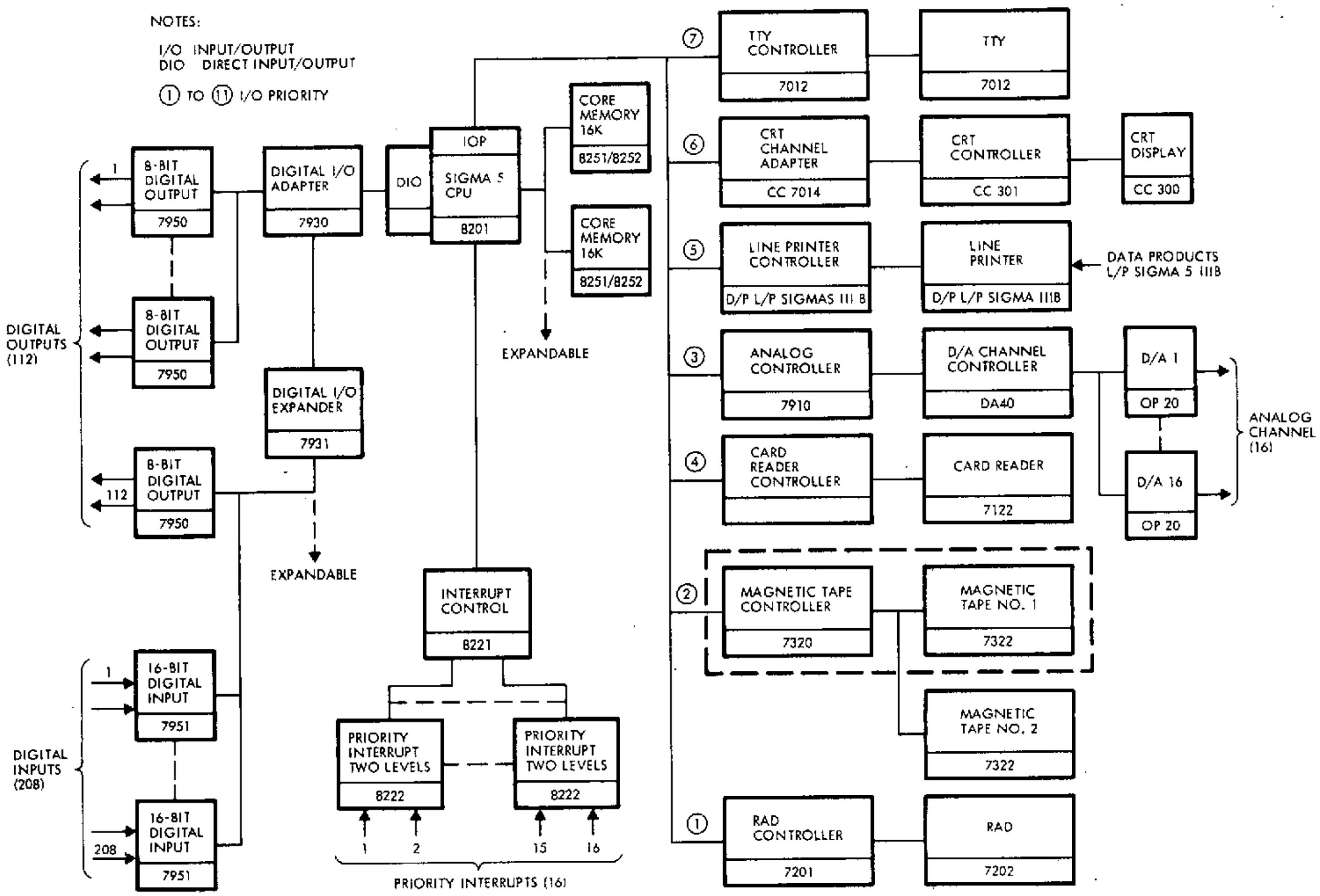


Figure 4-19. Pioneer ADPE Block Diagram

Table 4-5. Sigma 5 Equipment List

FUNCTION	MODEL	DESCRIPTION	EXTENDED PRICES: LIST PURCHASE PRICE	QUANTITY	PURCHASE PRICE	10 MS OEM DISCOUNT PRICE	MAINTENANCE CONTRACT RATE	FIXED TERMS LEASE RATES 1 YEAR	4 YEARS
COMPUTER	8201	SIGMA 5 CPU WITH INTEGRAL IOP	70 000	1	70 000	49 140	464	1 750	1 645
	8267	MEMORY BANK, 32K WORDS (32 BITS)	85 000	1	85 000	59 670	474	3 650	3 432
	8270	EXTERNAL INTERFACE FEATURE	2 000	1	2 000	1 404	10	50	47
	8221	INTERRUPT CONTROL CHASSIS	2 200	1	2 200	1 544	30	55	52
	8222	PRIORITY INTERRUPT, TWO LEVELS	350	8	2 800	1 966	N/C	72	72
KEYBOARD CONTROLLER	7012	KEYBOARD/PRINTER AND CONTROLLER (KSR-35)	6 000	1	6 000	4 637	48	150	141
CARD READER	7122	CARD READER, 400 CHARACTERS PER MINUTE	12 000	1	12 000	10 080	127	400	376
LINE PRINTER	7441	BUFFERED LINE PRINTER, 1100 LINES PER MINUTE	46 000	1	46 000	36 389	292	1 150	1 081
MAGNETIC TAPE SUBSYSTEM	7315	MAGNETIC TAPE CONTROLLER AND ONE DRIVE	16 000	1	16 000	13 440	200	600	550
	7316	ADD-ON TAPE DRIVE	12 000	1	12 000	10 080	180	450	420
DISC SUBSYSTEM	7250	CARTRIDGE DISC CONTROL	8 000	1	8 000	6 182	35	200	188
	7251	CARTRIDGE DISC (2.3 MB)	5 500	1	5 500	4 250	50	140	125
ANALOG OUTPUT SUBSYSTEM	7910	ANALOG OUTPUT CONTROLLER	4 500	1	4 500	3 478	23	113	107
	DA40	D/A CHANNEL CONTROLLER	1 000	1	1 000	840	20	110	50
	OP20	D/A CONVERTER	150	16	2 400	2 016	N/C	217	112
DISPLAY SUBSYSTEM	0033D	DISPLAY CONTROLLER, LOCAL, 80100/200 COMPATIBLE	6 000	1	6 000	6 000	30	195	163
	80100	DISPLAY CONTROLLER AND KEYBOARD 24 x 40 BLACK AND WHITE	900	1	900	900	12	51	33
	BV133	58.42 CM (23-INCH) BLACK AND WHITE STUDIO MONITOR	3 000	1	3 000	3 000	20	115	100
	7930	DIGITAL I/O ADAPTER	3 000	1	3 000	2 318	15	75	70
	7931	DIGITAL I/O EXPANDER	2 000	1	2 000	1 545	10		
DIGITAL I/O SUBSYSTEM	7950	STORED OUTPUT MODULE (8 BITS)	100	14	1 400	1 081	N/C	336	336
	7951	DIGITAL INPUT MODULE (16 BITS)	85	13	935	720	N/C	352	352
SUBTOTALS						211 680	2 040	11 156	10 329
	YT19	SYSTEMS CABINET, 48.26-CM (19-INCH) RAILS	1 200	1	1 200	948	N/C	33	32
ANALOG OUTPUT SUBSYSTEM	OP26	CONTROL/INDICATOR DOOR	300	1	300	252	N/C	27	8
INTERFACING & PACKAGING	OP27	SIU INTERFACE	320	1	320	269	N/C	29	8
	ET 10-15	DAC — SIU CABLES	70	2	140	114	N/C	13	5
	ET 13	CONTROL CABLES	20	4	80	64	N/C	7	3
TOTALS (1)					294 375	213 327	2 040	11 265	10 385

- The telemetry processor interfaces with the telemetry data console and performs preprocessing and data compression of the telemetry data. The telemetry processor tasks may be functionally divided as follows:

  Verification of telemetry word and frame synchronization

  Management of the data base

  Decommutation of engineering and scientific instrument words

  Determination of limit-checked words and notification of out-of-limit of improper engineering data.

- The science processor is functionally devoted to processing required for the scientific instruments.

The 64K words of core memory illustrated in Figure 4-20 is divided as follows:

- Telemetry and system controller shared memory, 16K words
- Science and system controller shared memory, 16K words
- Telemetry processor private memory, 16K words
- Science processor private memory, 16K words.

The shared memory allows each processor on the memory bus to access the memory as if it were its own. This capability enhances the multiprocessing capabilities among the three processors.

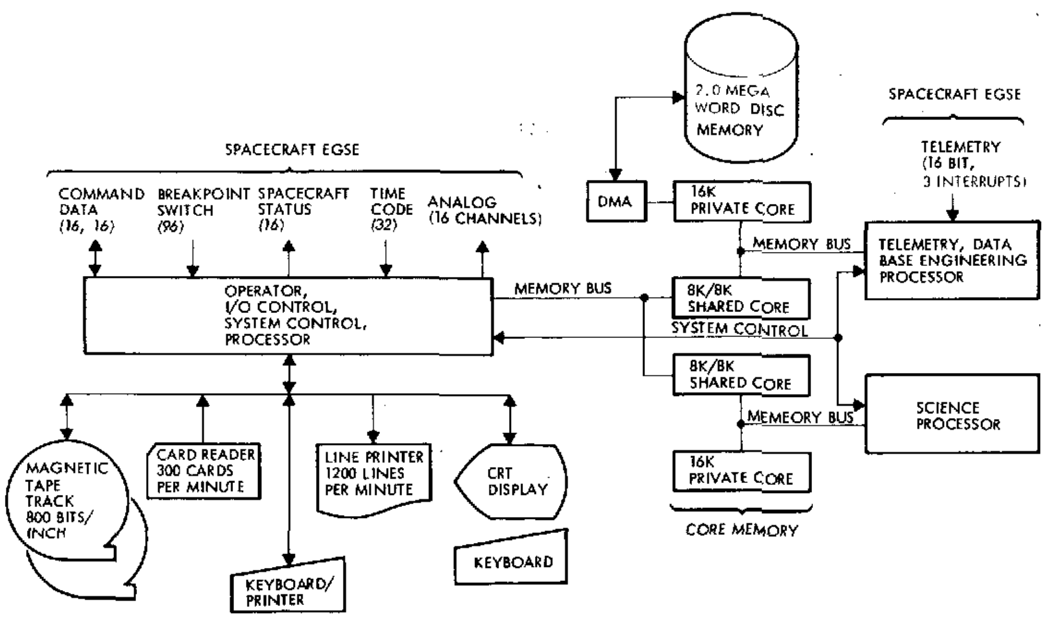


Figure 4-20. Pioneer Venus ADPE Distributed System Configuration

The peripheral and the EGSE input/output for the distributed system are sized to meet the requirements detailed in Section 2 of this study.

A distributed system was configured using the Varian 73 processor as a representative system. Table 4-6 lists the equipment and associated prices. The main features of this system are detailed below:

- CPU (each processor)

  Core memory capacity: 64K bytes expandable to 512K bytes

  Memory cycle time: 660 nanoseconds

  Hardware multiply divide

  Priority interrupts: up to 64 levels of priority interrupt

  Direct memory access input/output bus

  Power failure/restart

  Dual memory bus

- Peripheral

  Typewriter — TI 733 console, 30 characters/second input/output rate

  Magnetic tape (2) — 9-track, 800 bits/s, 25 in./s, industry compatible

  Line printer — 1200 lines per minute, 132 columns

  Card reader — 200 cards per minute

  CRT display — 20/40 characters with associated keyboard

  Disc memory — 2.34 16-bit words, 20 m/s average access time, 92K word per second transfer rate

- EGSE interface

  Digital input/output, priority interrupts — each processor is tailored to contain the required (see Section 4.4.2.1, EGSE Interfaces) input/output; interfaces are TTL compatible

  Analog — 16 channel, 10 bits, 10 volt.

4.4.4 Tradeoff Evaluation

A tradeoff evaluation for implementing the Pioneer Venus ADPE with a Sigma 5 (Pioneers 10 and 11) configuration versus a distributed

Table 4-6. Pioneer Venus ADPE Parts List and Prices for Three-CPU Distributed System

	MODEL NUMBER	ITEM	QTY	PRICE
SYSTEM CONTROL PROCESSOR				
1.	7001	PROCESSOR	(1)	$ 11 000
2.	7024	MEMORY	(4)	20 000
3.	7962	EXPERIMENT CHASSIS	(1)	250
4.	7966	BACKPLANE WIRING, RIGHT HAND	(1)	500
5.	7967	BACKPLANE WIRING, LEFT HAND	(1)	500
6.	7160	PRIORITY INTERRUPT MODULE (PIM)	(4)	2 000
7.	7910	DUAL CONTROLLER ADAPTER	(1)	300
8.	7920	INPUT/OUTPUT CABLE	(1)	200
9.	7955	POWER SUPPLY	(1)	1 000
10.	620-88	ANALOG POWER SUPPLY	(1)	495
11.	620-830A	DIGITAL OUTPUT MODULE	(2)	1 400
12.	620-831A	DIGITAL INPUT MODULE	(1)	1 000
13.	620-831B	DIGITAL INPUT EXPANSION MODULE	(2)	1 400
14.	620-32	MAGNETIC TAPE UNIT AND CONTROLLER	(1)	9 000
15.	620-32A	MAGNETIC TAPE UNIT SLAVE	(1)	7 000
16.	E-2119B	DATA PRODUCTS LINE PRINTER AND CONT.	(1)	28 200
17.	620-25	CARD READER AND CONTROLLER	(1)	4 000
18.	E-2250E	CATHODE RAY TUBE DISPLAY	(1)	5 400
19.	E-3002	TI 733 CONSOLE	(1)	5 400
20.	620-20	BUFFER INTERLACE CONTROLLER (BIC)	(2)	1 000
21.	620-870	DIGITAL/ANALOG CONTROLLER	(2)	1 590
22.	620-870B	TWO, 10-BIT DIGITAL/ANALOG	(7)	4 765
				$106 400
TELEMETRY PROCESSOR				
1.	7001	PROCESSOR	(1)	$ 11 000
2.	7024	MEMORY	(2)	10 000
3.	7962	EXPANSION CHASSIS	(1)	250
4.	7966	BACKPLANE WIRING, RIGHT HAND	(1)	500
5.	7960	MEMORY EXPANSION CHASSIS	(1)	1 000
6.	7921	MEMORY EXPANSION CABLE	(1)	200
7.	7160	PRIORITY INTERRUPT MODULE (PIM)	(2)	1 000
8.	7910	DUAL CONTROLLER ADAPTER	(1)	300
9.	7920	INPUT/OUTPUT CABLE	(1)	200
10.	7955	POWER SUPPLY	(1)	1 000
11.	7950	POWER SUPPLY	(1)	2 000
12.	620-830A	DIGITAL OUTPUT MODULE	(1)	700
13.	720-20	BUFFER INTERLACE CONTROLLER (BIC)	(2)	1 000
14.	620-36D	DISC MEMORY AND CONTROLLER	(1)	12 500
				$41 650
SCIENCE PROCESSOR				
1.	7000	PROCESSOR	(1)	$10 000
2.	7024	MEMORY	(2)	20 000
3.	7960	MEMORY EXPANSION CHASSIS	(1)	1 000
4.	7921	MEMORY EXPANSION CABLE	(1)	200
5.	7160	PRIORITY INTERRUPT MODULE (PIM)	(2)	1 000
6.	7910	DUAL CONTROLLER ADAPTER	(1)	300
7.	7911	SYSTEM CONTROLLER ADAPTER (SCA)	(1)	150
8.	620-830A	DIGITAL OUTPUT MODULE	(1)	700
				$33 350
		SYSTEM INTEGRATION		$ 5 000
		SYSTEM CONTROL PROCESSOR		106 400
		TELEMETRY PROCESSOR		41 650
		SCIENCE PROCESSOR		33 350
		CABINETS (2)		2 200
		GRAND TOTAL		$188 600
		APPROXIMATELY 10 PERCENT DISCOUNT AVAILABLE)		
		MAINTENANCE		$ 1 035/ MONTH

minicomputer system will be presented. The broader question of evaluating the tradeoff of the centralized approach versus the distributed approach will not be detailed, since there are obvious savings of specifying the Sigma 5 as the processor for the centralized configuration. These savings arise in the software and EGSE hardware commonality that may be derived by using the Pioneers 10 and 11 ADPE configuration.

4.4.4.1 System Analysis

A list of the advantages and disadvantages of each of the two concepts is presented below:

Sigma 5 System

Advantages:

- Proven data handling and performance for Pioneers 10 and 11
- Proven highly reliable
- Excellent field services for equipment maintenance
- All peripheral supported by one company service organization
- Wide variety of hardware and vendor provided software
- Superior operating system
- Operational software from Pioneers 10 and 11 applicable to Pioneer Venus
- Interface compatible with proposed EGSE
- TRW and NASA/ARC experience in Sigma 5 processing
- Possible government use of equipment after completion of spacecraft launch operations.

Disadvantages:

- Higher purchase cost
- Higher maintenance cost
- Greater facility requirements
- More difficult to modify or expand
- Computer life probably less than the more contemporary minicomputer.

Distributed Processor

Advantages:

- Represents approach using the latest technology
- Lower purchase cost
- Lower maintenance service cost
- Minimum facility requirements
- Highly flexible and easily accommodates changes and expanding requirements
- Provides extremely high rates and parallel multiprocessing.

Disadvantages:

- Requires entirely new set of operational software
- Requires some modification to EGSE
- Requires additional software to provide internal director tasks
- Performance and software have not been proven as applied to real time spacecraft testing
- Maintenance services questionable among various manufacturers.

4.4.4.2 Cost Analysis

Those areas that have a cost impact between the two approaches may be divided into the following categories: systems engineering, software development, and purchase and maintenance. These items, along with a cost summary, will be presented in the following paragraphs.

System Engineering

The Sigma 5 configuration would require a minimal amount of system engineering since the system is already designed. The distributed system, however, requires such items as a selection study, new specifications, new test procedures, and manufacturing. The estimated additional cost increase of the distributed system over the Sigma 5 Pioneer configuration due to system engineering is shown below:

Task	Cost Estimate ($K)
Selection study	10.0
New computer specification	4.0
EGSE validation software	12.0
Procurement liaison	2.5
EGSE interface design	7.5
EGSE interface manufacturing	3.0
	$41.0K

The $41.0K would not be a recurring cost if a second ADPE were required.

Software

Comparison may be made between the Sigma 5 and the distributed system on the basis of total programming costs, assuming the Sigma 5 program has no carryover from past efforts.

Program Segment	Program Cost	
	Sigma 5	Distributed
Operating System	Vendor-supplied	Vendor-supplied
Internal Director	--	25 percent more programming (estimated at $20K)
Engineering	Same	Same
Science	Same	Same

The increased cost for the internal director is attributable to the additional programming necessary to delegate tasks and control the task order and performance in the multicomputer environment.

If we consider the Sigma 5 Pioneers 10 and 11 software to be composed of roughly two segments of equal cost (science and engineering processing), then an estimate may be generated for software that may be reasonable. It is estimated that little, if any, of the science experiment processing is recoverable beyond the general area of control methods, since the experiments are not the same, in general. Therefore, recoverable software will be limited to engineering processing.

The engineering processing may be broken into the following areas:

1) Telemetry input processor
2) Breakpoint sample and control (CBKPTSMP)
3) Executive controller (CXSKED)
4) Telemetry initialization (CINIT)
5) Engineering mainframe processor (CENGMFP)
6) Summary print processor (CSUMRY)
7) CRT header update (CFIXUA)
8) Special function processor (CSPIDT)
9) CRT initialization (CRTSETUP)
10) Calfile generation (CALFIL)

The first four listed above will be so extensively changed that they might better be rewritten. It is estimated that they comprise 25 percent of the engineering processing. Of the remaining 75 percent (Items 5 through 10 above), the percentage recoverable is estimated as:

	Percent
CENGMFP	90
CSUMRY	90
CFIXUP	70
CSPIDT	75
CRTSETUP	80
CALFIL	80

The overall recovery for these, with weighing for relative size, is estimated at 80 percent.

If the total cost of Pioneers 10 and 11 software was $580K, then the engineering processor portion would be $290K. The estimated 25 percent loss for items 2) through 4) above would be $72.5K, leaving the estimated possible recoverable at $217.5K. The estimated 80 percent recovery rate would then give a dollar total of $174K.

The total additional nonrecurring software cost for the distributed system including the additional internal director software is estimated at $194K.

Purchase and Maintenance Costs

The cost and data presented in Tables 4-5 and 4-6 will be used to estimate the relative purchase and maintenance costs of the two concepts. Since the need period (July 1975 to August 1978) is 3 years, in both systems purchase is cost effective versus leasing. The purchase cost comparisons are clouded by the fact that different minicomputer manufacturers vary on the O.E.M. or quantity discount rate. Varian has indicated a quantity discount of approximately 10 percent. Others may vary between no discount and 30 percent. The O.E.M. discount for TRW with XDS is indicated at approximately 27 percent. Due to the uncertainty of the minicomputer discounts, two cost comparisons will be presented, one using the indicated price lists and a second with expected discounts. The following table presents a cost comparison.

	Sigma 5		Distributed Processor	
	Price List	Discounted (27%)	Price List	Discounted (10%)
Purchase	294.3	213.3	188.6	170.6
G&A, MH (27%)	79.3	57.5	50.9	46.1
Maintenance (3 years)	73.4	73.4	37.3	37.3
G&A (19%)	13.9	13.9	7.1	7.1
TOTAL	$460.9K	$358.1K	$283.9K	$261.1K

The estimated savings of the distributed system for purchase and maintenance from the above table would be from $97.0K (discounted) to $177.0K (list prices).

Cost Summary

The following is a cost summary indicating the savings or cost increase of the Pioneers 10 and 11 Sigma 5 versus the distributed:

Systems engineering (savings)	$41.0K
Software (savings)	$194.0K
Purchase and maintenance costs (increase)	<$97.0K - $177.0K>
Total savings using the Sigma 5	$58.0K - 138.0K

4.4.4.3 Conclusions

The Sigma 5 system is the most cost-effective approach and it is recommended that it be used for the Pioneer Venus ADPE. It has demonstrated and proven on the Pioneers 10 and 11 programs that it can successfully meet the Pioneer Venus spacecraft test requirement. Product support of the system throughout the needed time frame is guaranteed by XDS. It is felt the distributed processor is a viable feasible approach; however, it is a new concept to spacecraft testing and yet to be proven.

Further enhancement to the recommendation of the Sigma 5 as the ADPE is the fact that NASA/ARC uses a Sigma 5 for the Pioneers 10 and 11 ground station. The commonality between the Pioneer Venus and Pioneers 10 and 11 would provide a considerable software development savings. Further savings could also be obtained if the Sigma 5 is GFE. The Sigma 5 is probably more available than the newer minicomputers.

Since both TRW and NASA/ARC have personnel experienced in operational system development and operations using the Sigma 5, it should prove to be a highly cost-effective approach.

4.5 EGSE DEVELOPMENT APPROACHES

The following paragraphs describe some of the management techniques that are proposed that will minimize costs:

- Drawing control. All engineering drawings will comply with G-level standards.
- Specifications, test procedures. Modifications of existing test procedures and specifications will comply with the existing standards. New specifications and test procedures will be written to the short form format.
- Manufacturing technique. Manufacturing will build to nonreleased red-line drawings.
- Inspection. Inspection will be done to release prints and limited to workmanship only. This will be at the drawer level and console levels. Those units modified will be inspected as-built. Components that are to be sealed will be inspected only at the time of coverup.
- Parts. Parts procured will meet with the best commercial standards and be TRW approved. Parts procurement will not require customer approval.
- Acceptance test. Final buyoffs will be based on function performance only, and witnessed by quality assurance personnel. Customer or DCAS witnesses will not be required.
- Documentation. No formal technical manuals will be required for TRW design units.

4.6 EGSE EQUIPMENT LIST

This section presents a summary of the equipment identified during the Pioneer Venus study. Table 4-7 indicates the current plans for the following information for each item of EGSE: 1) utilization, 2) location of use, 3) quantity required, 4) design status, and 5) equipment source. The utilization columns indicate the various ground operations that each item of EGSE supports. The fact that many items of EGSE support several operations is advantageous from the standpoint of minimizing the amount of EGSE required (and thereby minimizing the cost of the EGSE) and also provides better correlation of test data between the various operations.

The equipment source columns indicate that most of the EGSE is available from the Pioneers 10 and 11 program TRW and MMC capital equipment. There is some launch site data communications equipment listed that should be obtained from CKAFC as GFE.

Table 4-7. Pioneer Venus Support Equipment Requirements

SUPPORT EQUIPMENT NOMENCLATURE	UTILIZATION											USED AT							DESIGN STATUS				EQUIPMENT SOURCE						
	DESIGN TESTING	INTEGRATION SUPPORT	INTEGRATION TESTING	QUALIFICATION TEST	ACCEPTANCE TEST	COMPATIBILITY TEST	SERVICING	HANDLING	TRANSPORTATION	LAUNCH SUPPORT	IN PROCESS TESTING	VENDOR	MMC DENVER	MMC ORLANDO	TRW	LAUNCH SITE	OTHER	QUANTITY	PROGRAM DERIVATION	AS IS	MODIFIED	NEW	SOURCE	MODIFY	NEW	BUY	GFE	CAPITAL	DIRECT CHARGE
RADIO FREQUENCY CONSOLE	X		X	X	X	X				X					X	X		1	PIONEER		X		PIONEER	X					
COMMAND TRANSMITTER NO. 1																		1	10&11		X		10&11	X			X		
COMMAND TRANSMITTER NO. 2																		1			X			X			X		
FREQUENCY COUNTER																		1		X							X		
TELEMETRY RECEIVER																		1			X			X			X		
COMMAND ENCODER																		1		X							X		
RF MONITOR																		1		X							X		
RAMP GENERATOR																		1		X							X		
COMMAND TRANSMITTER POWER SUPPLY																		1		X							X		
CONSOLE POWER CONTROL																		1		X							X		
COMMAND ENCODER SELECTOR																		1		X							X		
INTERCOM																		1		X			TRW					X	
RAPID COMMAND INTERFACE																		1		X			PIONEER 10&11				X		
TIME DISPLAY																		1		X			TRW					X	
OSCILLOSCOPE																		1		X			PIONEER 10&11				X		
ANTENNA ASSEMBLY																		1		X							X		
RECORDER CONSOLE	X		X	X	X	X				X					X	X		1			X			X					
PREAMPLIFIERS (SANBORN)																		1		X							X		
POWER CONTROL (SANBORN)																		1		X							X		
SANBORN STRIP CHART RECORDER																		1		X							X		
MAGNETIC TAPE RECORDER																		1		X							X		
INSTRUMENTATION PATCH PANEL																		1			X			X			X		
CONNECTOR PANEL																		1		X							X		
CONSOLE POWER CONTROL																		1		X							X		
INTERCOMMUNICATIONS, FOUR CHANNEL																		1		X			TRW					X	
POWER CONTROL (TAPE RECORDER)																		1		X			PIONEER 10&11				X		
ELECTRONICS (TAPE RECORDER)																		1		X							X		
GROUND CONTROL CONSOLE	X		X	X	X					X					X	X		1			X			X					
TEST SIGNAL INTERFACE UNIT																		1			X			X			X		
ACS SIMULATION AND CONTROL																		1			X			X			X		
INTERCOM																		1		X			TRW					X	
POWER MONITOR																		1			X		PIONEER 10&11	X			X		
DIGITAL VOLTMETER (VOLTAGE																		1		X			TRW					X	
DVM (CURRENT)																		1		X			PIONEER 10&11				X		
ORDNANCE CONTROL AND MONITOR																		1			X				X				X
CONSOLE POWER SUPPLY																		1		X			TRW					X	
ORDNANCE POWER SUPPLY																		1	CMCL	X			CMCL		X				X
PRIMARY POWER CONTROL																		1	PIONEER 10&11	X			PIONEER 10&11				X		
TIME DISPLAY																		1		X			TRW					X	
OSCILLOSCOPE																		1		X			TRW					X	
SOLAR ARRAY SIMULATOR																		1			X							X	
ORDNANCE LOAD SIMULATOR																		1	PIONEER 10&11		X		PIONEER 10&11	X			X		
THRUSTER SIMULATORS (DUAL)																		4			X			X			X		
SUN SENSOR STIMULUS																		1			X			X			X		
SOLAR ARRAY SIMULATOR POWER SUPPLY																		1	CMCL	X			CMCL		X				X
SPACECRAFT EXTERNAL POWER SUPPLY																		1	CMCL	X			CMCL		X				X

Table 4-7. Pioneer Venus Support Equipment Requirements (Continued)

SUPPORT EQUIPMENT NOMENCLATURE	UTILIZATION: DESIGN TESTING	INTEGRATION SUPPORT	INTEGRATION TESTING	QUALIFICATION TEST	ACCEPTANCE TEST	COMPATIBILITY TEST	SERVICING	HANDLING	TRANSPORTATION	LAUNCH SUPPORT	IN PROCESS TESTING	USED AT: VENDOR	MMC DENVER	MMC ORLANDO	TRW	LAUNCH SITE	OTHER	QUANTITY	DESIGN STATUS: PROGRAM DERIVATION	AS IS	MODIFIED	NEW	EQUIPMENT SOURCE: SOURCE	MODIFY	NEW	BUY	GFE	CAPITAL	DIRECT CHARGE
TELEMETRY DATA CONSOLE	X		X	X	X	X				X					X	X		1	PIONEER 10&11		X		PIONEER 10&11	X					
TIME CODE TRANSLATOR/GENERATOR																		1		X			TRW					X	
FIXED WORD DISPLAY																		1			X		PIONEER 10&11	X			X		
PSK DEMOD EMR 2726																		1		X			TRW					X	
DECODER/BUFFER																		1		X			PIONEER 10&11				X		
PCM BIT SYNC EMR 2720																		1		X			TRW					X	
SPACECRAFT STATUS DISPLAY																		1			X		PIONEER 10&11	X			X		
CONSOLE POWER CONTROL																		1		X			PIONEER 10&11				X		
PCM DECOMMUTATOR EMR 2746																		1		X			TRW					X	
DECIMAL DISPLAY EMR 2756																		1		X			PIONEER 10&11				X		
POWER SUPPLY +28 VDC																		1		X			PIONEER 10&11				X		
POWER SUPPLY -15 VDC																		1		X							X		
POWER SUPPLY +15 V																		1		X							X		
INTERCOM																		1		X			TRW					X	
MONITOR ASSEMBLY, POWER SUPPLY																		1		X			PIONEER 10&11				X		
DATA FORMAT GENERATOR	X		X	X	X	X				X					X	X		1		X									
MINICOMPUTER																		1		X							X		
PSK DATA MODULATOR																		1		X							X		
ASR-33																		1		X							X		
PRIMARY POWER CONTROL																		1		X							X		
SOFTWARE																					X								X
TEST CONDUCTOR CONSOLE	X		X	X	X					X					X	X		1		X									
OSCILLOSCOPE																		1		X			TRW					X	
SIGNAL MONITOR																		1		X			PIONEER 10&11				X		
INTERCOM																		2		X			TRW					X	
REMOTE TIME DISPLAY																		1		X			TRW					X	
TV DISPLAY SELECT																		1		X			PIONEER 10&11				X		
TV DISPLAY																		1		X							X		
DECIMAL DISPLAY EMR 2756																		1		X							X		
COMMAND ENCODER																		1		X							X		
PRIMARY POWER CONTROL																		1		X							X		
RECEIVER SIGNAL STRENGTH																		1		X							X		
PERIPHERAL EQUIPMENT																													
SERIES IN-LINE FUSE BOX	X		X	X	X	X									X			6		X							X		
STS POWER VALIDATOR	X		X	X	X	X				X					X	X		1				X			X				X
STS SIGNAL VALIDATOR	X		X	X	X	X				X					X	X		1			X			X			X		
FIELD INTENSITY METER SET	X		X	X		X									X			1		X							X		
CABLE SET	X		X	X	X	X				X					X	X		1			X			X			X		
SPACECRAFT ORDNANCE CHECKOUT UNIT	X		X	X	X	X				X					X	X		1	TRW	X			TRW					X	
INTERCOM (BLOCK HOUSE UNIT)										X								1	CKAFS	X			ETR				X		
SRM SAFE/ARM MONITOR AND CONTROL (BLOCK HOUSE UNIT)										X								1				X			X				X
POWER SUPPLY (BLOCK HOUSE UNIT)										X								1	CMCL	X			TRW					X	
PRIMARY POWER CONTROL (BLOCK HOUSE UNIT)										X								1	DSCSI	X			TRW					X	
LAUNCH SITE COMMUNICATIONS EQUIPMENT																													
HIGH-GAIN ANTENNA, S-BAND										X						X		2	CKAFS	X			ETR				X		
MEDIUM GAIN ANTENNA, S-BAND										X						X		1		X							X		
COAX LINES										X						X		2		X							X		
VIDEO AMPLIFIER										X						X		2		X							X		
VIDEO TRANSMISSION LINES										X						X		2		X							X		

Table 4-7. Pioneer Venus Support Equipment Requirements (Continued)

SUPPORT EQUIPMENT NOMENCLATURE	UTILIZATION											USED AT							DESIGN STATUS				EQUIPMENT SOURCE						
	DESIGN TESTING	INTEGRATION SUPPORT	INTEGRATION TESTING	QUALIFICATION TEST	ACCEPTANCE TEST	COMPATIBILITY TEST	SERVICING	HANDLING	TRANSPORTATION	LAUNCH SUPPORT	IN PROCESS TESTING	VENDOR	MMC DENVER	MMC ORLANDO	TRW	LAUNCH SITE	OTHER	QUANTITY	PROGRAM DERIVATION	AS IS	MODIFIED	NEW	SOURCE	MODIFY	NEW	BUY	GFE	CAPITAL	DIRECT CHARGE
AUTOMATIC DATA PROCESS EQUIPMENT	X		X	X	X	X				X					X	X		1	PIONEER 10&11	X			PIONEER 10&11						
COMPUTER CENTRAL PROCESSOR																		1		X							X		
BREAK POINT SWITCH UNIT																		1		X							X		
CRT DISPLAY																		1		X							X		
CARD READER																		1		X							X		
PAPER TAPE READER																		1		X							X		
LINE PRINTER																		1		X							X		
TELETYPEWRITER																		1		X							X		
KEY PUNCH																		1		X			TRW					X	
D/A CONVERTERS																		16		X			PIONEER 10&11				X		
DIGITAL MAGNETIC TAPE UNIT																		2		X							X		
DIGITAL INPUT/OUTPUT																		1		X							X		
RANDOM ACCESS DISC FILE																		1		X							X		
INTERCOM																		1		X			TRW					X	
PROBE RF TEST SET	X		X	X	X					X			X	X	X	X		1											
VARIABLE ATTENUATOR																		1				X			X				X
S-BAND TEST OSCILLATOR																		1				X			X				X
TEST/PATCH PANEL																		1				X			X				X
S-BAND RECEIVER																		1				X			X				X
PSK SUBCARRIER DEMODULATOR																		1				X			X				X
BIT SYNCHRONIZER AND A/D CONVERTER																		1				X			X				X
SERIAL-TO-PARALLEL BUFFER																		1				X			X				X
ANTENNA COUPLER																		1				X			X				X
BUS/PROBE INTERFACE SIMULATOR	X		X	X	X					X			X	X	X	X		1											
TEST/CONVERTER PANEL																		1				X			X				X
TIME CODE DISPLAY																		1				X			X				X
DATA CLOCK AND CONTROL																		1				X			X				X
COMMAND GENERATOR																		1				X			X				X
BATTERY CHARGE CONTROL																		1				X			X				X
POWER CONTROL																		1				X			X				X
BATTERY CHARGER																		1				X			X				X
POWER SUPPLY																		1	CMCL	X					X	X			X
PROBE PYRO SIMULATOR	X		X	X	X					X			X	X	X	X		1											
SAFE/ARM INDICATOR PANEL																		1				X			X				X
FIRE SIGNAL MONITOR																		1				X			X				X
PYRO SIMULATOR PANEL																		1				X			X				X
PROBE ORDNANCE TEST SET			X	X	X					X			X	X	X	X													
PYRO MONITOR																		1	MMC	X								X	

Table 4-7. Pioneer Venus Support Equipment Requirements (Continued)

SUPPORT EQUIPMENT NOMENCLATURE	UTILIZATION											USED AT							DESIGN STATUS				EQUIPMENT SOURCE						
	DESIGN TESTING	INTEGRATION SUPPORT	INTEGRATION TESTING	QUALIFICATION TEST	ACCEPTANCE TEST	COMPATIBILITY TEST	SERVICING	HANDLING	TRANSPORTATION	LAUNCH SUPPORT	IN PROCESS TESTING	VENDOR	MMC DENVER	MMC ORLANDO	TRW	LAUNCH SITE	OTHER	QUANTITY	PROGRAM DERIVATION	AS IS	MODIFIED	NEW	SOURCE	MODIFY	NEW	BUY	GFE	CAPITAL	DIRECT CHARGE
DATA PROCESSING EQUIPMENT	X		X		X	X							X	X				1	MMC	X									
COMPUTER CONTROL PROCESSOR																		1		X			MMC					X	
CRT DISPLAY																		1		X								X	
CARD READER																		1		X								X	
PAPER TAPE READER																		1		X								X	
LINE PRINTER																		1		X								X	
TELETYPEWRITER																		1		X								X	
D/A CONVERTERS																		1		X								X	
WIDEBAND ANALOG TAPE RECORDER																		1		X								X	
DIGITAL INPUT/OUTPUT																		1		X								X	
RANDOM ACCESS DISC FILE																		1		X								X	
DIRECT WRITE OSCILLOGRAPHS																		1		X								X	
SPACECRAFT EXPERIMENT INTERFACE SIMULATOR						X									X			1	PIONEER 10&11		X		PIONEER 10&11						
POWER SUPPLY DRAWER																		1			X			X			X		
POWER SIMULATOR																		1			X			X			X		
COMPUTER																		1		X							X		
BUFFER UNIT																		1			X						X		
PRIMARY POWER CONTROL																		1		X							X		
TEST SIGNAL GENERATOR																		1			X						X		
ROLL REFERENCE GENERATOR																		1		X							X		
ROLL REFERENCE/COMMAND SIMULATOR																		1		X							X		
DTU SIMULATOR																		1			X						X		
DTU SIMULATOR POWER SUPPLY																		1		X							X		
LINE PRINTER																		1		X							X		
TELETYPE																		1		X							X		
PAPER TAPE READER																		1		X							X		
CABLE SET																		1				X			X				X

5. MECHANICAL GROUND SUPPORT EQUIPMENT

5.1 INTRODUCTION

This section describes the mechanical ground support equipment (MGSE), which includes all the mechanical equipment required to support the spacecraft and its subassemblies through the entire cycle of assembly, integration, test, checkout, transportation, and launch. MGSE includes the equipment required to support assembly and disassembly, transportation, storage, and mass properties measurements of the small and large probes.

The probe bus/orbiter MGSE will use existing Pioneer 10 and 11 hardware and design concepts to reduce development costs. In addition, certain items of TRW-owned capital equipment (for example, the propellant and pressurant loading system) will be directly applicable to the Pioneer Venus program.

The probe MGSE will primarily be new equipment, although certain items will include off-the-shelf commercial equipment. The equipment to be developed for Pioneer Venus will consist basically of slings, stands, and containers, using known techniques and methods.

5.2 REQUIREMENTS

The basic requirements for the Pioneer Venus MGSE were determined by analyzing factory through launch operations. The MGSE must provide for:

- All handling operations such as hoisting, rotating, positioning, and mating operations related to the spacecraft and probes
- Convenient and safe access to the spacecraft at any of the work or test sites within TRW and the launch site
- Handling and installation of heavy spacecraft components, such as the orbit insertion motor
- Protection for all sensitive and fragile spacecraft surfaces and components
- Transportation of the spacecraft within the TRW facility, to the launch site, and between test areas at the launch site
- Protection for the spacecraft from adverse environments during transportation

- Measurement and recording of environmental conditions surrounding the spacecraft during transportation
- Leak testing and servicing for the reaction control equipment with propellant and pressurant
- Measurement and verification of all critical spacecraft alignments
- Support for the spacecraft during special tests such as probe, appendage and ram platform deployment, and thermal vacuum tests
- Support assembly and disassembly operations of the large and small probes.

For convenience the MGSE has been divided into the following functional groups:

- Ground handling equipment (GHE)
- Transporter and accessories
- Spacecraft shipping equipment
- Probe ground support equipment
- Leak test equipment
- Propulsion loading equipment
- Orbit insertion motor fixtures and test equipment
- Alignment equipment
- Ground support equipment
- Government-furnished equipment.

Most items support similar operations at the manufacturing plant and launch site, minimizing the amount of MGSE required. Several MGSE items will be provided by TRW as capital equipment and some major items exist as government-furnished equipment. The use of TRW-owned and government-owned equipment will help minimize Pioneer Venus costs.

Some of the MGSE will be new items designed specifically for Pioneer Venus. However, they will incorporate proven designs that have been used successfully on many previous projects. This will ensure maximum MGSE effectiveness and minimize compatibility problems between the MGSE and the spacecraft and/or facilities.

5.3 BUS/ORBITER, USE OF PIONEERS 10 AND 11

Some of the MGSE that was used on the Pioneers 10 and 11 program will be usable as-is in support of the Pioneer Venus spacecraft. The selection of the Atlas/Centaur launch vehicle has reduced the amount of Pioneers 10 and 11 equipment that had been planned for use if the Thor/Delta was selected as the launch vehicle. The Thor/Delta launch vehicle would have allowed unmodified utilization of the aft adapters, band clamps, and spacers.

The following items will be usable as-is or with some modification:

- Rotation fixture - assuming that the only operations for which this fixture is used are with the spacecraft without probes installed, no deboost motor installed, and without propellant (i.e., assumed spacecraft weights of 175 kilograms (385 pounds) (probe) and 242 kilograms (572 pounds) (orbiter), and spacecraft longitudinal center of gravity approximately the same distance from the separation plane as Pioneers 10 and 11, only modification to the spacecraft/rotation fixture spacer is required.

- Hoist sling - the hoist fittings that attach to the spacecraft will be redesigned to suit the hard points to be provided at the spacecraft frame T-members, and to provide adequate safety margins for the heavier Pioneer Venus spacecraft. A small amount of rework and redesign of the hoist sling corner plates will be required to support the increased loads.

- Magnetics test fixtures - the nonmagnetic track and dolly used in the Pioneers 10 and 11 program will be usable as-is in support of proposed magnetics testing for Pioneer Venus. Additional equipment will be required to provide environmental protection for the spacecraft and to lower and raise the degaussing coils around the spacecraft.

- Propellant loading unit - this capital equipment, developed for Pioneers 10 and 11, will be usable as-is.

5.4 NEW EQUIPMENT

Those integration and test activities which require new equipment are:

- Probes handling and test
- Deboost motor installation

This equipment is discussed in the following section.

5.4.1 Probe Ground Support Equipment (GSE)

The new probe GSE for Pioneer Venus is as follows:

- Small probe bench top assembly stands with integral shelf support ring and pressure shell support ring
- Small probe aeroshell stand with aeroshell support ring
- Large probe descent capsule stands with equipment ring support ring, tension cone support ring, and aft aeroshell support ring
- Large probe forward aeroshell support stand and ring
- Large probe and small probe shipping container
- Sling set
- Weight and center of gravity adapters.

Figures 5-1 and 5-2 depict the ground support equipment used in the normal assembly sequence for both the large and small probes. The following describes the probe GSE:

- The small probe bench top assembly stands consist of a tubular structure with two trunions to allow rotation of the appropriate support ring. A floor-mounted configuration was considered instead of the bench mount; however, because these stands are small, the bench-mounted configuration was selected. Both stands are made the same size, even though the one used for the integral shelf could be shorter and narrower, to eliminate the need for two sets of equipment, thus allowing greater use and flexibility. The integral shelf support ring and pressure shell support rings are of different diameter and configuration to interface with the appropriate probe unit; however, a common bench stand/ring interface is used.
- The small probe aeroshell stand and the large probe descent capsule stands are the same design. The selected configuration is a modified, off-the-shelf automobile engine stand. The commercial stand modification includes providing new rings and adapter plates, adding stabilizing jacks, and replacing casters and the ring stand lubrication system to be compatible with cleanliness requirements. A new design stand was considered but the modified commercial stand was selected to minimize costs. Mounted on this common stand, the rings interface with the appropriate probe unit; then can be rotated and locked, thus affording access to all areas of the probe.

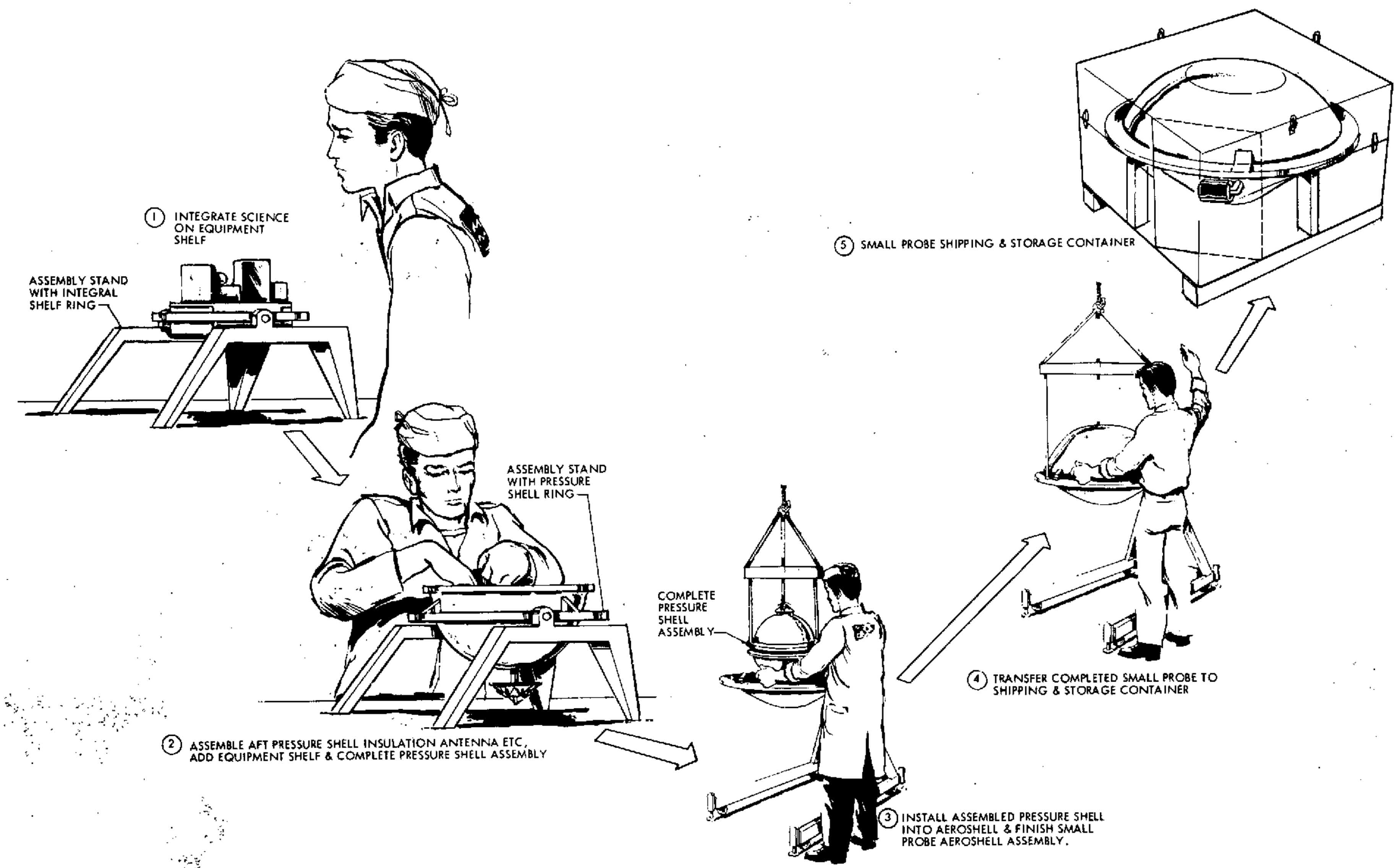


Figure 5-1. Small Probe Assembly and Handling Equipment

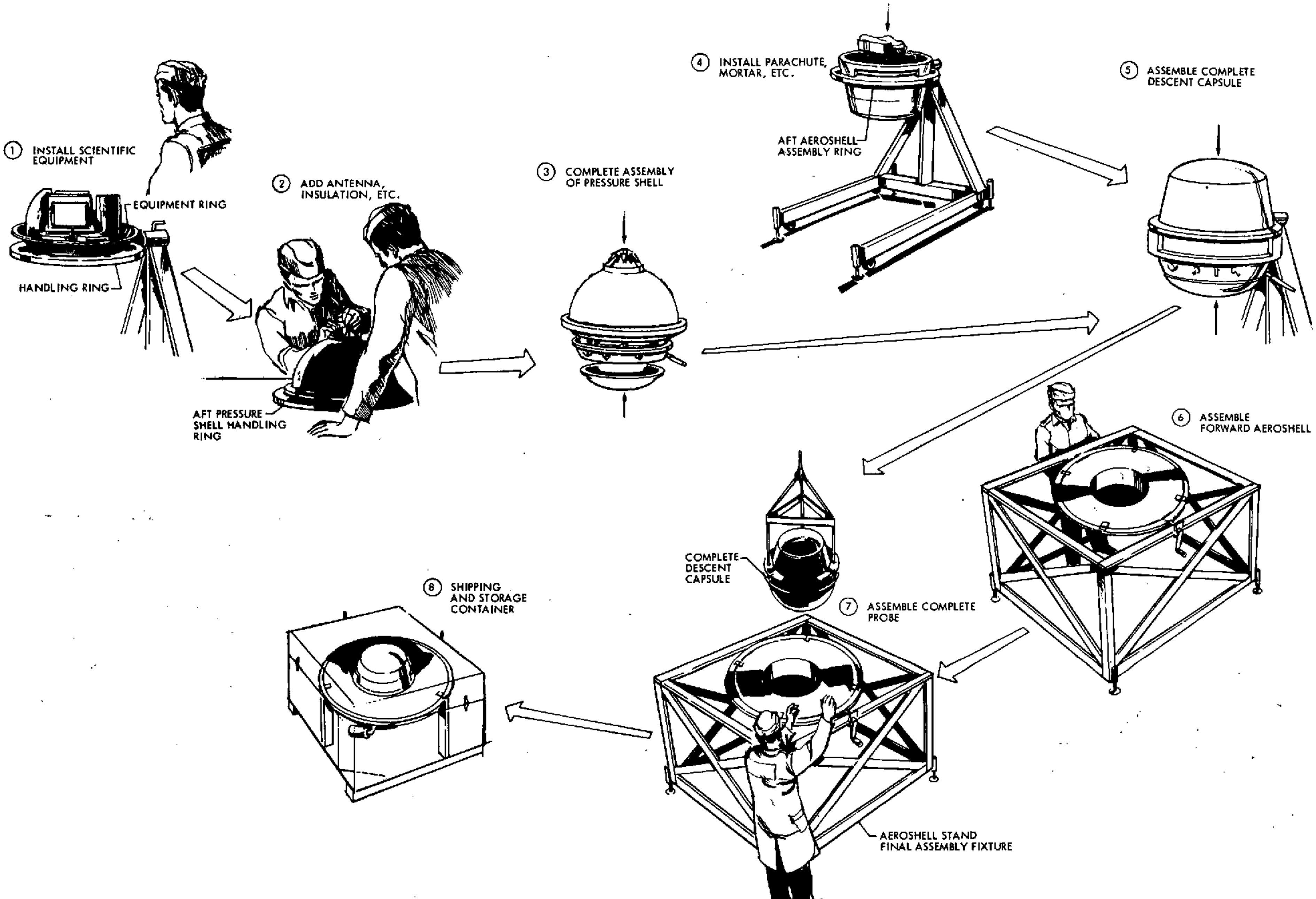


Figure 5-2. Large Probe Assembly and Handling Equipment

- The large probe forward aeroshell support stand is a four-legged design that supports the aeroshell in a ring at a convenient height above the floor. The stand can be rotated via two trunions and a manual rotational drive and lock. The ring, similar to the small probe aeroshell support ring, supports the aeroshell about its circumference on a padded, beveled surface that allows the aeroshell to nest in the ring. A series of padded clamp segments, which extend over the top of the aeroshell, secure the aeroshell to the ring for positive restraint.

- The shipping and storage container is constructed of wood with a fiberglass skin to minimize costs. The aeroshell handling ring is bolted to the container frame for support. No shock isolation system or temperature control system is provided because special handling will be required.

- The sling set utilizes off-the-shelf cables, hardware, and fittings.

- The weight and center of gravity adapters provide a method of support for the probes and probe subassemblies for mass properties measurements and balance operations.

5.4.2 Probe Deployment Testing

The large probe deployment tests can be performed by setting the spacecraft on an aft adapter and attaching the probe through a sling to a bungee cord, and counterweighted to provide simulated zero-g condition deployment. A gimbal or four-bar linkage could give the necessary degree of freedom so that tipoff could be observed.

The test can be performed using a spin table and instrumenting through slip rings. However, the advantages of a spinning deployment test for this probe do not appear to be sufficiently meaningful that it should be preferred over a less complicated nonspinning test.

The small probes present more of a problem since they are deployed in a plane normal to the spacecraft spin axis. If it is a requirement that the test would simulate in-flight conditions by deploying from a spinning spacecraft, then the problems to be overcome would be:

- Support of the probe to eliminate the local gravity effect

- Some means of arresting and catching the probes as they are released from the spacecraft.

It is assumed however, that the major purpose of such a deployment test would not be to demonstrate predictable dynamic effects, but would be to demonstrate that the restrain mechanism releases properly and that there is no interference or hangup as the probe moves away under a known force. In this case, the test would seem to be simplified in that it could be done with the spacecraft stationary, with the probe supported to eliminate one-g effects, and an external force applied through a cable attached to the probe to simulate separation.

A spinning test could be accomplished by attaching a platform to the spacecraft to support the probes in a plane normal to the spacecraft spin axis. A tether or similar snagging device would restrain the probe within the limits of the platform. The test could be performed using a spin table and instrumenting through slip rings. Alternatively, a spinning test could be performed in which the probe was supported by an overhead cable to eliminate gravity effects. This test would require the cable also to be attached to a rotating system with a capability of following the probe path at release. This latter capability would be required to eliminate the forces imposed by a fixed cable relative to a moving probe.

Another alternative test fixture is one in which the probe is supported in a yoke attached to a rack and pinion gear arrangement. The pinion gear is connected to the spin table so that motion of the probe can be controlled as a function of spacecraft rotation. This test fixture has the advantage of being able to demonstrate that clearances exist with the probes and spacecraft in motion relative to each other without the associated problems of arresting a moving probe. An added advantage is that clearance throughout the whole separation cycle can be observed; these clearances cannot be observed with a spinning spacecraft and unrestrained probes.

It is recognized that the aerodynamic effects of a spinning spacecraft may be unpredictable in such a test, and the possibility exists that a real deployment problem could be manifested that would be nonexistent in the absence of such effects. Due to these uncertainties and the increased cost of a spinning test, the better approach is to perform a nonspinning test as described above.

5.4.3 Magnetometer Boom Deployment

This test will be similar to the boom deployment performed on Pioneers 10 and 11. The boom will be deployed across a smooth surface, supported by either a low friction bearing or air bearing.

5.4.4 Deboost Motor Installation

For this installation, it is proposed that the motor be set on a support stand, nozzle down, and the spacecraft lowered onto the motor. This is a reverse of the method used on Intelsat III (where the motors were lowered into an inverted spacecraft) and eliminates the requirement for an expensive piece of MGSE, unique to this operation, to invert the spacecraft.

5.4.5 Probe Handling for Spacecraft Integration

For the large probe, the most feasible arrangement is a clamp which fits around the probe large diameter and can be attached to an overhead sling. For the small probe, the following alternative approaches are possible for handling and installing into the spacecraft:

- A C-clamp type attaching fixture that holds the probe top and bottom
- A clamp that fits around the probe large diameter and is attached to an overhead sling. This would be used in conjunction with a vertically adjustable track and dolly. The probe would be moved in and out of the spacecraft/probe clamp on the dolly.
- A vertically adjustable mobile stand with a hinged cradle to support the probe.

5.4.6 Mass Properties Equipment

A new fixture, which can support the spacecraft with the spin axis vertical and is adaptable to existing TRW capital equipment, is required for mass properties measurements. An alternative approach considered was to use existing Pioneers 10 and 11 mass properties hardware. This approach was rejected, based on a determination of its cost and the ease of handling operations involved in measurements made with the spin axis horizontal. This determination was based on the following requirements:

- Modifying the rotation fixture to support the Pioneer Venus with probes and motor installed
- Designing and fabricating new hardware to be used in conjuntion with existing Pioneers 10 and 11 mass properties hardware.

5.5 MGSE DEVELOPMENT APPROACH

5.5.1 General MGSE Test Philosophy and Methods of Verification

A combination of analysis, inspection, demonstration, and formal tests will verify that the MGSE meets all requirements. The final verification test will be the actual use of the MGSE with the structural model or qualification model. Emphasis will be on achieving a high level of confidence in the functional and safety aspects of the requirements at minimum cost. Since none of the MGSE items involve advancement of the state of the art or new technology, little or no development testing will be required. The designs will be conservative with high safety margins and will, in all cases, utilize technical principles which are well understood and have been extensively verified on previous programs at TRW. The methods by which various requirements will be verified are summarized below:

- Verification by analysis. An analysis study will be performed on certain characteristics of the MGSE that are impractical or impossible to demonstrate, such as reliability, maintainability, failure modes and analysis, and thermal characteristics of the thermal vacuum fixture and the shipping container.

- Verification by inspection. Throughout the manufacturing and acceptance testing phases, quality assurance will verify that the items are in conformance to overall drawing dimensions and requirements, markings and identifications, color and finishes, and corrosion protection.

- Verification by demonstration. Verification by demonstration is illustrating the fit and functional characteristics of the MGSE that must be physically demonstrated to show conformance to requirements. These items include interface fit checks, mobility demonstrations and human performance trials.

- Verification by test. Testing includes weighing each major type, proof load tests to structural items, proof pressure tests to pressurized items to ensure personnel safety, performance tolerance tests, and dynamic load attentuation tests for transporting equipment.

Details of the major tests are:

a) Proof load tests. All critical load-carrying MGSE structures will be proof tested to 2 times the maximum expected load. The items that will be proof-load tested include all hoistings and handling equipment and all transportation tie-down fittings.

b) Proof pressure tests. All pressurized containers will be proof-pressure tested to 2 times the maximum expected pressure.

c) Performance tolerance tests. Equipment that must perform within a specific tolerance with flight units will be tested to demonstrate that the required accuracy has been achieved. The accuracy test will be conducted with flight configuration hardware if there is no resultant safety hazard.

5.5.2 Testing and Inspection of MGSE

A composite procedure will be prepared covering MGSE ground handling equipment items. The procedure will include a list of equipment to be tested, objectives, data requirements, type of tests for each type of MGSE test loads, test setup sketch, and general test procedure.

5.5.3 MGSE Maintenance

MGSE components will use either standard, off-the-shelf equipment where possible, or be developed and fabricated in-house from raw stock material. In either case, rapid replacement or repair of any part is planned. Use of critical, long-lead time material or components will be avoided. Quick interim repairs can be performed immediately from TRW supplies of raw stock and standard hardware; therefore, no formal MGSE spares list is anticipated.

To support testing in multiple locations, more than one of several items will be built. This provides a redundancy that can be used in an emergency.

5.5.4 Drawing Requirements

The rules that apply to MGSE drawings are intended to be as flexible as possible within the existing drafting standards. MGSE product design drawings will meet the following requirements:

- The most liberal tolerances allowable will be utilized. Whenever possible, one-place decimal dimensions will be used. This will automatically define the tolerance as plus or minus 1/10 of an inch. All tighter tolerances, which are called out, must be justified to the responsible MGSE engineer.

- The TRW Drafting Room Manual will be followed for general format, drawing control, and release.

5.5.5 Fabrication

If appropriate, the fabrication and assembly of MGSE end items will be accomplished in the MGSE Shop. This shop will be supported as required by the larger shops to provide the most effective and efficient methods when these needs are identified.

The MGSE Shop will reduce to a minimum any operations considered to be costmetic in nature. The shop will be concerned primarily with quality, low cost, and commercial practice fabrication techniques.

Manufacturing will work in close coordination with the MGSE engineer to ensure short-line method of communication. A build-to-print philosophy will be employed, relying upon the specialized skills of the shop technician to determine the optimum operational sequence.

6. PERFORMANCE ASSURANCE

This section describes how TRW Systems will implement a performance assurance program that is cost effective, assures compliance with high reliability standards, and is consistent with Pioneer Venus project requirements. The performance assurance effort during the study was primarily concerned with supporting the system and subsystem design tradeoffs in terms of reliability analysis for the various options. The low-cost approach in the reliability and quality assurance areas was studied in terms of confidence in the system reliability versus cost. The results of this study indicated that the approach used for Pioneers 10 and 11 is already very cost effective, compared to other programs, and hence it is anticipated that the only modification needed for the Pioneer Venus performance assurance effort will be detailed cost saving improvements.

The performance assurance functions will include the following disciplines: reliability; safety; parts, materials and processes; quality assurance; and configuration management. The APM for performance assurance reports directly to the Pioneer Venus project manager and has overall responsibility for the performance assurance program tasks (Figure 6-1). The reliability/safety manager; parts, materials and processes (PMP) manager; quality assurance manager; and configuration management manager report to the APM for performance assurance. This type of organization assures the resources necessary to objectively evaluate problems, and recommend and pursue effective solutions.

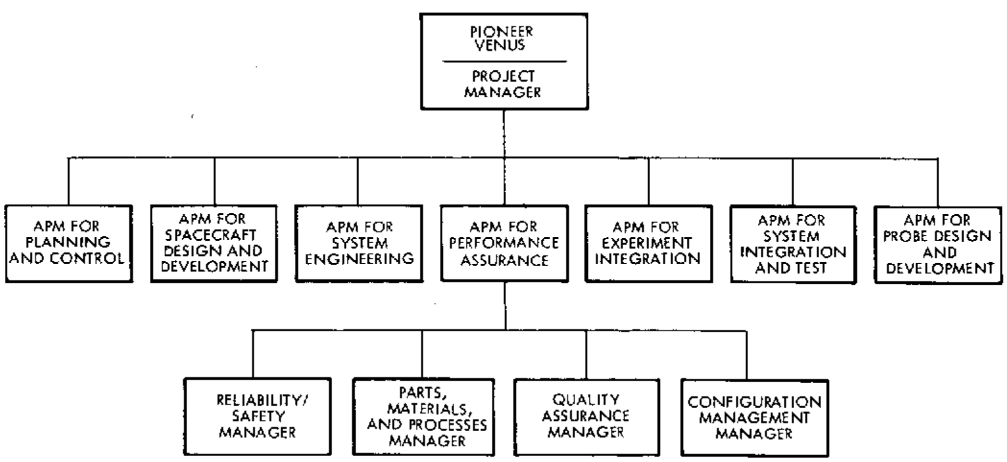


Figure 6-1. Performance Assurance Organization

The performance assurance program assures that high reliability is inherent in the design, development, and manufacturing phases and is proven during the integrated test program. This achievement of design reliability is accomplished by:

- A thorough design review program to assure proper design practices
- Use of proven procedures, standards, and controls
- Use of space-qualified parts and flight-proven designs
- Implementation of a comprehensive test program
- Use of a closed-loop failure reporting, analysis, and corrective action system.

6.1 RELIABILITY/SAFETY

The reliability program will be performed in accordance with a plan to be prepared and submitted for approval to ARC. The program will be based on NHB 5300.4(1A) and include those tasks considered necessary and cost effective in achieving Pioneer Venus project objectives. In general, the plan will implement a program very similar to that utilized on Pioneers 10 and 11. In conjunction with the reliability effort, the safety aspects of the spacecraft design will be considered in the reliability analyses.

6.1.1 Design Review

A design review program will be implemented by performance assurance personnel to assure that performance and reliability requirements are met by a satisfactory and cost-effective design. Maximum commonality in the design of the orbiter, probe bus, and probes is one of the design goals and this commonality will be utilized to the maximum practical extent to achieve a cost-effective program, from initial design through manufacturing, test, spacecraft integration, and launch activities.

Design reviews on the spacecraft systems will be performed at various scheduled project milestones. The objective of system reviews will be to evaluate system requirements, capabilities, and design status, and to identify problem areas and proposed resolutions. The system-level design reviews will include an initial project requirements review on the

multiprobe and orbiter spacecraft systems. These system reviews will be time-phased with appropriate unit/subsystem design reviews and tests on engineering models, qualification units, probe subsystem, spacecraft structures, and spacecraft thermal models.

The design review committee will include cognizant personnel from all appropriate disciplines. The committee members will be selected for their technical ability, maturity, objectiveness, and independence to assure a thorough evaluation and criticism of the design. Where practical, senior design specialists will be selected from organizations not directly responsible for the design being reviewed to assure independent criticism. Minutes of the design review meeting, action items assigned, and approved responses will be included in a design review closeout document.

NASA/ARC will be invited to attend all design reviews and will receive design review data packages in advance. MMC personnel will participate in the probe bus system and interface design reviews as appropriate.

6.1.2 Reliability Analyses

Beginning early in the design phase, reliability analyses will be initiated to provide timely identification of potential reliability problems and to assist in design tradeoff studies. As the design progresses and details become more evident, reliability analyses will be refined and and updated to assure continuation of sound reliability practices. The results of these analyses will be included with appropriate design review data. Pioneers 10 and 11 reliability analyses of unchanged designs will be used to the maximum feasible extent in the overall analyses effort.

6.1.2.1 Reliability Apportionment and Prediction

A reliability goal for both the orbiter and probe bus spacecraft will be apportioned to the subsystems and units during the initial design phase and updated periodically to reflect the prevailing design. Reliability apportionment to the subsystems and units will aid in serving as a forcing function for design discipline. Reliability predictions will be used in design tradeoff studies, in redundancy studies, and in probability of occurrence determinations of critical items and single-point failures.

With these analyses, an optimum spacecraft configuration based on reliability, cost, performance, and other critical system parameters will be established.

6.1.2.2 Failure Mode and Effects Analyses

Failure mode and effects analyses (FMEA) will be performed during the design phase to identify modes of failure and their effects on mission objectives. Particular attention will be given to performing interface FMEA between units. The purpose will be to identify and reconcile potential failures that may be mission catastrophic, inadvertently couple one unit to another, or cause failure of redundant units. The interface analyses performed on Pioneers 10 and 11 proved beneficial and indicated a need for additional consideration and effort for the Pioneer Venus project.

The orbiter and multiprobe spacecraft designs will be critically reviewed to eliminate single-point failures within cost and practicality constraints. These potential failure points will be identified with the appropriate FMEA and the action taken for resolution will be described in the design review data packages.

6.1.2.3 Hazard Analyses

Safety hazard analyses will be performed in conjunction with the FMEA and will be refined as the spacecraft designs progress. These analyses will provide the basis for:

- Initiating design changes to eliminate or minimize safety hazards
- Instituting safety devices and safeguards where appropriate
- Identifying safety actions required during hazardous time periods
- Recommending special procedures for servicing, handling, storage, and transportation of the spacecraft.

6.1.3 Failure Reporting and Corrective Action

The established TRW failure correction system tailored to meet project requirements will be used on Pioneer Venus project equipment. It will be a closed-loop system for reporting failures, analyzing the failures, and issuing corrective action items as was implemented on Pioneers 10 and 11.

A joint ARC/TRW/MMC failure review board meeting will be convened approximately monthly to review failures, failure analyses, and corrective actions, and to approve failure closeout. The members of the board for TRW and MMC will include representation from the performance assurance, reliability, quality assurance, failure correction, test, and design engineering disciplines as appropriate. ARC representation will include the spacecraft manager, reliability and quality assurance manager, and the in-plant representative(s). Minutes of the meeting will include the action taken on each failure, action items assigned, and agreements reached.

6.1.4 Subcontractor Control

Reliability controls and requirements will be placed upon each of the subcontractors and/or suppliers in accordance with product complexity, function, and degree of criticality. Each subcontractor and supplier will be monitored for performance by the reliability manager in accordance with the applicable requirements established by a subcontractor reliability requirements control document and referenced in the subcontract or purchase order.

The ARC reliability requirements imposed on TRW will in turn be imposed on MMC via a subcontractor reliability requirements document. MMC will be required to prepare a TRW-approved reliability program plan based on these requirements.

6.1.5 Parts, Materials, and Processes

A high reliability parts, materials and processes program will be implemented on Pioneer Venus as was accomplished on Pioneers 10 and 11. This high-reliability and cost-effective program will be achieved by:

- Using only space-qualified parts and materials
- Using residual Pioneers 10 and 11 parts to the maximum extent
- Assuring maximum use of existing parts and materials specifications for new procurements
- Providing design support to achieve design goals and high reliability with maximum parts commonality

- Assuring that part screening and burn-in is performed in accordance with specification and Pioneer Venus project requirements
- Providing support to manufacturing to assure proper process control and problem correction
- Performing failure analyses on parts and materials to identify failure mechanism and recommend corrective action.

The MMC parts, materials and processes program will be specified in the subcontractor reliability requirements document.

6.2 QUALITY ASSURANCE

The quality assurance (QA) program will fulfill the essential requirements of NHB 5300.4(1B), "Quality Program Provisions for Aeronautical and Space System Contractors." MMC will be required to implement a TRW-approved quality assurance program plan to the same ARC requirements as those imposed on TRW. TRW will impose these requirements via the subcontractor quality program requirements document. The QA program plan will implement the controls necessary to ensure that the Pioneer Venus spacecraft designs are not compromised during the manufacturing cycle nor jeopardized during the test, integration, and launch phases.

6.2.1 Procurement Control

TRW QA personnel will be responsible for assuring the adequacy and quality of all purchased articles, materials, parts, components, processes, and services. These requirements will be specified in appropriate subcontractor and supplier quality requirements documents.

QA personnel will participate in evaluations of subcontractors and suppliers to ensure that their quality system will adequately meet the project quality requirements. This evaluation is based on review of recent supplier quality history, supplier quality surveys, and review of the suppliers' quality programs and inspection plans for comparable in-house programs.

Purchase orders, subcontracts, and associated specifications for the procurement of articles or services will be reviewed prior to release to assure imposition of appropriate quality requirements.

Depending upon the item being procured, TRW will assign a QA representative to act as a resident or itinerant at the major subcontractors and/or supplier facilities.

6.2.2 Fabrication Controls

Manufacturing documentation will be reviewed by quality planning personnel to verify that proper identification of hardware will be maintained throughout all assembly, test, and processing activities. Quality planning personnel will verify that the requirements specified on the engineering drawings are compatible with planning and that operational sequences of manufacturing and/or assembly allow for adequate inspection points. Special inspection requirements will be inserted into the planning as necessary.

The TRW Process Requirements Specifications (PR), Quality Operation Instructions (QOI's), and Fabrication/Inspection Process Procedures (FIPP's) will define the acceptance criteria to be used in manufacturing and inspection of the hardware. Existing Pioneer 10 and 11 standards will be used to the maximum extent possible.

6.2.3 Inspection and Tests

In-process inspection and test operations will be established at appropriate intervals during the fabrication and process operations to verify the article's conformance to drawing and specification requirements. Visual aids, inspection checklists, general inspection instructions, and integrated planning documents will be used to effect adequate inspection before the last point at which acceptability of the operation or quality of the characteristic may be verified.

QA personnel will provide inspection observation of tests conducted on qualification and deliverable units during manufacturing, qualification, and acceptance tests. These observations will assure that the test is conducted in accordance with an approved test procedure, test equipment setup, and calibration requirements.

Spacecraft/probe assembly, integration, and test operations will be conducted under QA observation. Quality assurance personnel will conduct final visual inspections, configuration identification inspections, and test observation throughout the integration and test cycle.

6.2.4 Nonconforming Material Control

To segregate discrepant items, controlled areas will be established in receiving inspection, manufacturing, assembly, and test facilities. Nonconforming material detected during the receiving, fabrication, refurbishment, assembly and test inspection operations will be identified, segregated, and withheld from use. When segregation is not feasible of is physically impossible, the item will be "bonded in place" and held for material review. A material review board, similar to the one on Pioneers 10 and 11, will assign the disposition of discrepant hardware. Procedures governing the activities of the board will be delineated in the Pioneer Venus Quality Assurance Plan.

6.3 CONFIGURATION MANAGEMENT

The configuration management office (CMO) will develop, implement, and maintain a configuration management system (CMS) based on minimum documentation and controls to assure a cost-effective program. The CMS assures that all contract end items are properly identified by engineering data and that changes to these data are accounted for, controlled, and verified.

The configuration management effort is consistent with the level of control needed for each phase of development, manufacture, and test. Initially, this effort concentrates on the system and interface specifications, orienting project personnel with the details of the CMS, and developing detail policy and implementation instructions. As the design and development phase continues, the configuration management effort will include control of equipment specifications beginning with format and consistency through the control of changes and status.

Engineering data (i.e., specifications, drawings, and test procedures) are released through the CMO to TRW's Configuration Administration and Data Management (CADM) group. All changes to engineering data are reviewed by the CMO. Class I changes are presented to a formal change control board for evaluation and submitted to NASA/ARC for approval.

All testing of qualification and flight hardware will be accomplished with the use of formal test procedures coordinated by and released through the CMO.

Manufacturing data will be encoded, enabling a computerized comparison of required design to as-built configuration for each serial numbered unit (black box).

The CMO provides continuing support at the launch site by identifying changes required for compatibility with GSE and launch site equipment. The CMS will assure that the exact in-flight configuration is properly documented.

6.3.1 Subcontractor Control

A program of subcontractor configuration management will be established early in the Pioneer Venus project by reviewing proposals and purchase orders for adequate configuration management requirements. TRW will impose a configuration management requirements document on all subcontractors, and will require audits of subcontractors configuration management practices as necessary.

6.4 REFURBISHMENT PROGRAM

6.4.1 Refurbishment of Residual Units

To achieve low cost and continued high reliability, the Pioneer Venus orbiter and probe bus spacecraft will utilize residual Pioneers 10 and 11 equipment wherever possible. Much of this space-qualified equipment can be used in its present condition with only minimum retest needed to verify its operation after storage.

Some of the equipment will require modifications to meet Pioneer Venus spacecraft performance requirements. In addition, some of the same equipment will require incorporation of outstanding engineering orders (EO) and/or replacement of nonflight parts. Many of the Pioneer 10 and 11 prototype and qualification units, under an authorized limited usage (ALU) system, were allowed to use nonflight parts or be used without the latest EO's because of schedule and cost constraints at that time. Table 6-1 gives a list of Pioneers 10 and 11 residual equipment planned for use on the Pioneer Venus spacecraft program with the major ALU's outstanding against the Pioneers 10 and 11 qualification and prototype equipment.

Table 6-1. Pioneer 10 and 11 Residual Equipment with Major ALU's Outstanding

UNIT	S/N	ALU NO.	ACTION REQUIRED
PCU	001	FL-0340-L160	REPLACE POTTER PT4-1062 CAPACITORS
	002	FL-0340-L160	REPLACE POTTER PT4-1062 CAPACITORS
INVERTER	001	FL-0360-L159	REPLACE POTTER PT4-1062 CAPACITORS
	002	FL-0360-L159	REPLACE POTTER PT4-1062 CAPACITORS
	003	FL-0360-L159	REPLACE POTTER PT4-1062 CAPACITORS
CTRF	002	FL-0301-P91	INCORPORATE PREVIOUS DESIGN CHANGES
CDU	001	FL-0210-L73	REPLACE PT4-4153 INTEGRATED CIRCUITS WHICH HAVE THIN METALIZATION
	001	FL-0210-L156	REPLACE POTTER PT4-1062 CAPACITORS
	002	FL-0210-L73	REPLACE PT4-4153 INTEGRATED CIRCUITS WHICH HAVE THIN METALIZATION
	002	FL-0210-L156	REPLACE POTTER PT4-1062 CAPACITORS
CEA	001	FL-0410-L72	REPLACE PT4-4153 INTEGRATED CIRCUITS WHICH HAVE THIN METALIZATION
	002	FL-0410-L72	REPLACE PT4-4153 INTEGRATED CIRCUITS WHICH HAVE THIN METALIZATION
DDU	001	FL-0501-L69	REPLACE PT4-4153 INTEGRATED CIRCUITS WHICH HAVE THIN METALIZATION
	002	FL-0501-L69	REPLACE PT4-4153 INTEGRATED CIRCUITS WHICH HAVE THIN METALIZATION
CSP	001	FL-0170-L71	REPLACE PT4-4153 INTEGRATED CIRCUITS WHICH HAVE THIN METALIZATION
	002	FL-0170-L71	REPLACE PT4-4153 INTEGRATED CIRCUITS WHICH HAVE THIN METALIZATION
TRANSMITTER DRIVER	002	FL-0130-P54	REPLACE JOHANSON PT4-1077/3-01 CAPACITORS
	002	FL-0130-P87	INCORPORATE PREVIOUS DESIGN CHANGES
	003	FL-0130-P54	REPLACE JOHANSON PT4-1077/3-01 CAPACITORS
	003	FL-0130-P87	INCORPORATE PREVIOUS DESIGN CHANGES
RECEIVER	001	FL-0140-L158	REPLACE POTTER PT4-1062 CAPACITORS
	002	FL-0140-P84	INCORPORATE PREVIOUS DESIGN CHANGES
	002	FL-0140-P157	REPLACE POTTER PT4-1062 CAPACITORS
	003	FL-0141-P85	INCORPORATE PREVIOUS DESIGN CHANGES
	003	FL-0141-P157	REPLACE POTTER PT4-1062 CAPACITORS
DTU	001	FL-0501-Q109	INCORPORATE PREVIOUS DESIGN CHANGES
	001	FL-0501-L117	INCORPORATE PREVIOUS DESIGN CHANGES
	001	FL-0501-L129	INCORPORATE PREVIOUS DESIGN CHANGES
	002	FL-0501-Q109	INCORPORATE PREVIOUS DESIGN CHANGES
	002	FL-0501-L117	INCORPORATE PREVIOUS DESIGN CHANGES
	002	FL-0501-L129	INCORPORATE PREVIOUS DESIGN CHANGES

6.4.2 Parts Availability

The Pioneer Venus spacecraft will utilize the same parts and/or part types as used on Pioneers 10 and 11 to the maximum extent possible. This approach will hold part qualification costs to a minimum. New designs will give preference to existing Pioneer 10 and 11 approved parts.

Approximately 50 percent of the parts required for the Pioneer Venus program are presently in Pioneers 10 and 11 stock. Of the parts requiring a repurchase order, it is anticipated that only four or five parts would not be available due to line discontinuance. In most cases, a new part type with better characteristics would be available. If necessary, a part satisfying the old characteristics would be purchased with special screening procedures. New part procurement specifications will be kept to a minimum. This will aid in achieving a low cost parts program.

APPENDIX A

PRELIMINARY PIONEER VENUS SCIENTIFIC INSTRUMENT INTERFACE AND INTEGRATION MANAGEMENT PLAN

1. PURPOSE AND INTRODUCTION

The scope of the scientific instrument interface and integration activities on the Pioneer Venus project includes the definition of instrument-related system interfaces; the definition of instrument to probe, probe bus, and orbiter spacecraft design interfaces; and the instrument-related integration and test activities. The objective of the scientific instrument interface and integration activities is to merge each of the instrument packages with the appropriate spacecraft vehicle (the four probes, probe bus, and orbiter) so as to permit all experiments to accomplish their scientific objectives. The purpose of this plan is to define the organization, responsibilities, tasks, methods, and documentation that will be employed by the spacecraft and probe contractor to carry out these activities in support of NASA/ARC, and includes the pertinent activities to be performed at Martin Marietta, Denver, under subcontract to TRW for the probes.

The GFE scientific instruments for the probes will be delivered, integrated, and tested with the probe vehicles in Denver. The GFE scientific instruments for the probe bus spacecraft, as well as the completely integrated probes, will be delivered to TRW in Redondo Beach, California for spacecraft integration and systems test, in which the probes will be treated essentially as additional spacecraft subsystems. Probe and probe bus scientific instruments will be tested as part of the spacecraft integrated systems test program.

In the case of the orbiter mission, if the European Space Research Organization (ESRO) participates with NASA in the mission, the orbiter spacecraft and all GFE scientific instruments will be delivered for complete integration and testing at the ESRO facility in Holland; if ESRO does not participate, instrument delivery and complete spacecraft integration and testing will occur at the TRW facility in Redondo Beach.

In every case, for both the probe and orbiter missions, after completion of spacecraft integration and test, the completely integrated

spacecraft will be delivered to CKAFS ETR for the prelaunch and launch operations, as provided in the launch operations program.

The general approach for management of scientific instrument interface and integration activities for Pioneer Venus is based upon the methods successfully used by ARC and TRW in managing these activities on Pioneers 6 through 11, with some modifications to take account of special Pioneer Venus circumstances. Consideration is given to the pertinent activities and schedule requirements at ESRO in case the orbiter mission is performed as a cooperative venture. Additional innovations are incorporated to reduce the costs and increase the effectiveness of the scientific instrument integration program without sacrificing mission reliability or scientific objectives, and to take advantage of previous experience in handling potential problems in the science integration activities of the previous Pioneer spacecraft.

2. MANAGEMENT OF SCIENTIFIC INSTRUMENT-RELATED ACTIVITIES

Figure A-1 is a conceptual diagram of experiment-related interface and integration activities, arranged to show the interrelations as they apply to the Pioneer Venus probe and orbiter missions. Activities which apply to both missions are shown in the center vertical column, with horizontal arrow lines to the left and the right from the common spacecraft and probe activity boxes to show how they support each stage of specific spacecraft vehicle and probe vehicle activity. These activities are shown in the left and right sides of the diagram, respectively.

Overall project system activities, shown schematically at the top of the diagram, can be subdivided into system engineering and system interface definition and implementation. System activities relating to the scientific experiments consist of determining those scientific requirements which have an impact on the spacecraft and probe system designs, then evaluating such requirements, and verifying or instituting the capability in the appropriate system design. Scientific instrument design interfaces with the appropriate vehicle (large or small probe, probe bus, or orbiter spacecraft) will be defined under ARC control. Interface problems will be resolved in coordination with ARC/experimenters, and rapid, effective

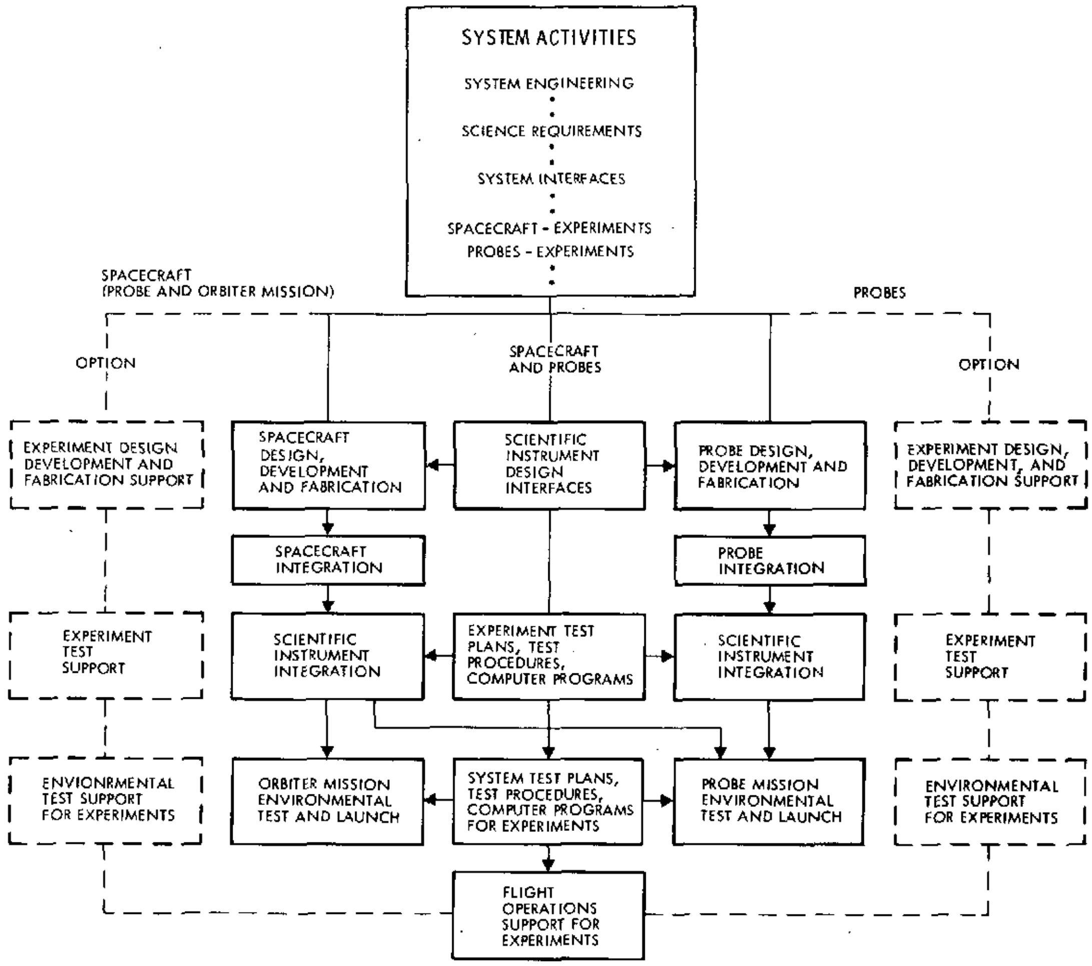


Figure A-1. Conceptual Diagram of Experiment-Related Interface and Integration Activities

interface change procedures will be established. These activities are described in Section 3. Scientific instrument integration and test activities are described in Section 4.

In this plan, scientific instrument integration and test activities include the necessary preparatory or pre-integration tasks which precede the physical integration of instruments, probes, and spacecraft. Preparation of experiment test plans, test procedures, and computer programs for use with experiment GSE are pre-integration activities to support scientific instrument integration. Generally, with only small modification these evolve into experiment system test plans, test procedures, and computer programs to support the integrated spacecraft system level environmental testing and pre-launch activities.

Figure A-1 also shows, by means of dashed lines, an option for the contractor to supply personnel for scientific instrument development and test under ARC direction for either or both spacecraft (probe bus and orbiter) instruments and probe instruments. This option is presented in an addendum to this appendix.

A summary of the tasks, organization, and preliminary schedules to carry out these activities is given in the following subsections.

2.1 Task Summary

An overall schematic diagram of the spacecraft and probe contractor's scientific instrument interface and integration tasks on the Pioneer Venus program is shown in Figure A-2. The tasks are grouped into categories of interface definition activities, instrument integration and test activities, and the associated documentation activities — both the documentation to be generated by the contractor and submitted to ARC and that received from ARC for review and/or implementation. Selected milestone events are indicated across the top of the figure so that the sequence of boxes from left to right constitutes a rough work flow diagram. These are general tasks, and the diagram is applicable to both probe and orbiter missions; within the probe mission, it includes both probe instrument and probe bus instrument tasks, as indicated by the term "Probes and Spacecraft" in the appropriate titles. For the orbiter mission, this term is understood to be "Spacecraft" alone.

A detailed breakdown of individual specific tasks for probe instruments, probe bus instruments, and orbiter instruments, with preliminary time schedules for each, is given in Section 2.3 below. The system and design interface activities and the integration and test activities are described in Sections 3 and 4, respectively.

The difference in terminology of instrument and experiment should be noted. As indicated above, scientific instruments are physically integrated with the appropriate probe and spacecraft vehicles in order to perform experiments to achieve the scientific objectives. This plan is concerned with scientific instrument interface and integration activities. In some cases, however, instrument considerations involve the broader concepts of experiment performance; in at least one case, the proposed

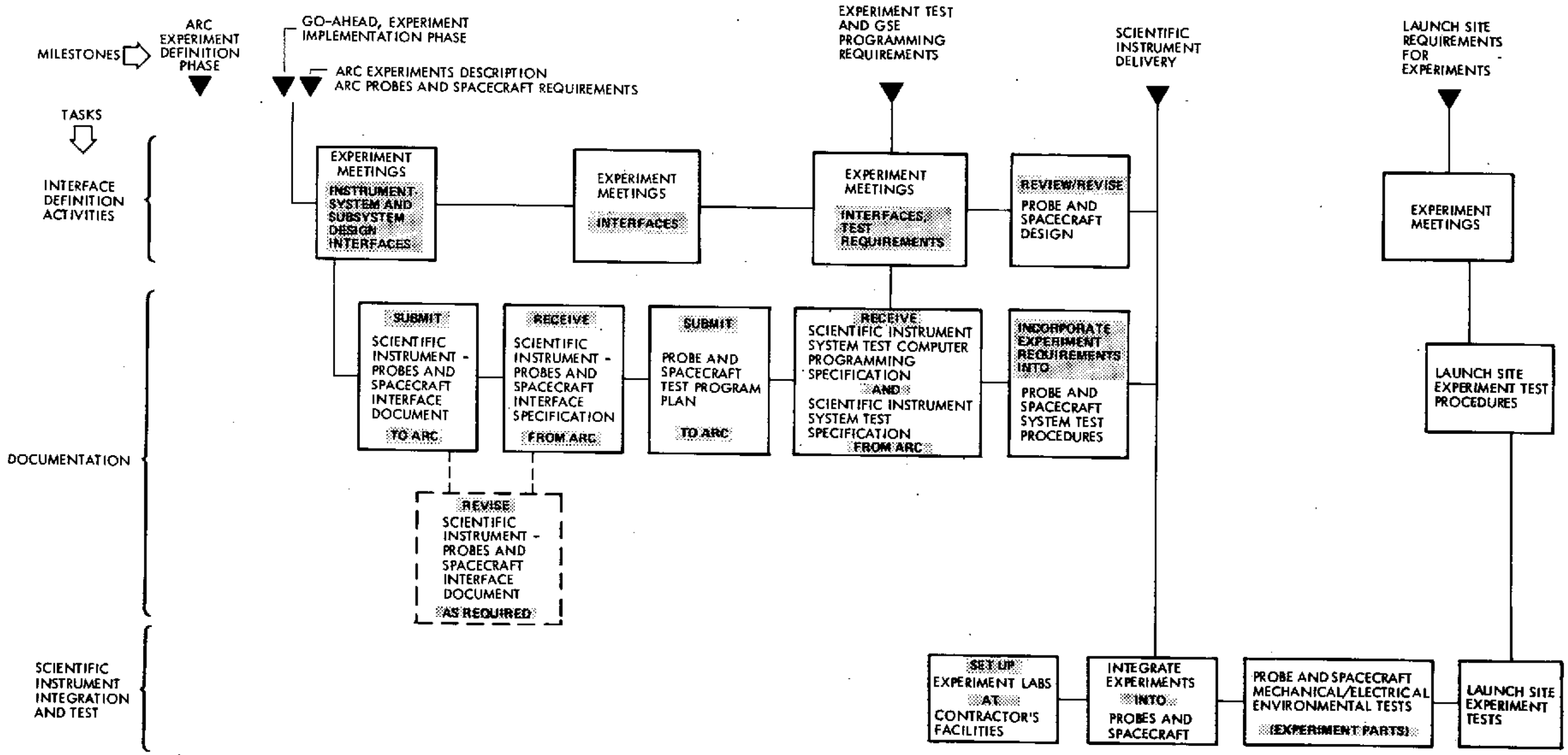


Figure A-2. Scientific Instrument Interface and Integration Tasks Overall Schematic Diagram

RF occultation experiment, a separate instrument package is not involved, although the experiment has system interface and test requirements. This plan uses the terms instrument and experiment as is most appropriate.

2.2 Organization

The Pioneer Venus scientific instrument interface and integration activities are under the direction of an assistant project manager (APM) for experiments, who reports directly to the Pioneer Venus project manager. The consolidation of all these activities under an experienced, experiment-oriented manager provides a single focus of responsibility for all experiment-related activities and for liaison with ARC and experiment personnel throughout the project. This procedure has been proven to be highly successful and cost effective on previous scientific spacecraft projects such as OGO and Pioneers 6 through 11. The APM for experiments will have a small, full-time staff of experiment engineers who will support the interface, integration, and test activities described in detail in Sections 3 and 4. The contractor organization for both spacecraft and probe activities is shown in Figure A-3.

For the spacecraft experiments (probe bus and orbiter), assistance will be provided by the various subsystem and system design groups, as indicated in Figure A-3, to establish the detailed design interfaces. TRW has found that it is much more cost effective to utilize these groups for

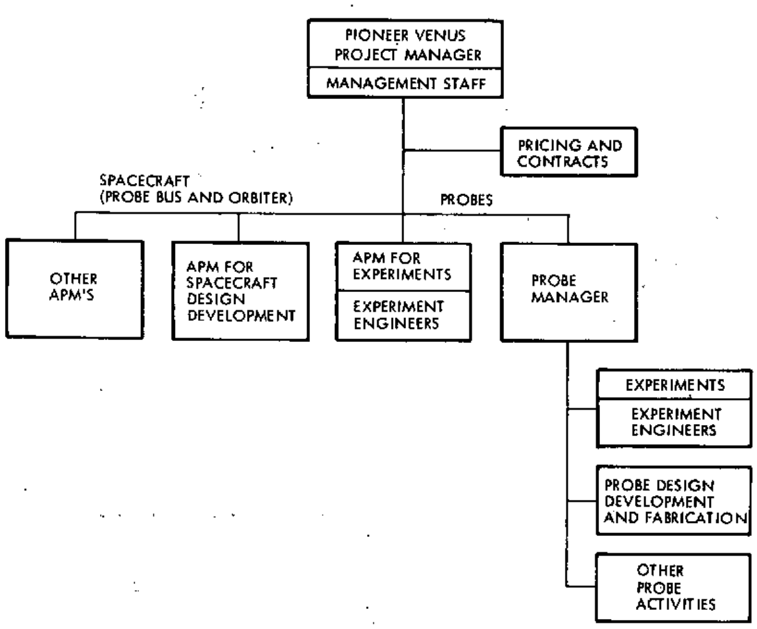


Figure A-3. Spacecraft and Probe Contractor Organization

interface design than to establish a separate design group, since it avoids needless duplication of effort and ensures prompt evaluation and implementation of all experiment requirements. This procedure also permits utilization of these subsystem and system experts only for the time required to support the detailed experiment activities, while project liaison and coordination continuity are provided by the experiment APM and his staff. The experiment engineers who work in the design coordination and interface activities from the start of the project will become part of the experiment test crew when the scientific instruments are delivered to TRW (or ESRO). This again takes full advantage of the experience and knowledge gained during the early phases of the project.

For the probe experiments, a similar organization will be utilized. A group of scientists/engineers which has been engaged in Pioneer Venus probe activities over the past 2 years at Martin Marietta will continue with the probe instrument interface and integration activities throughout the experiment implementation phase. Because of the large number of probe experiments, and the complexity and variety of the instrument-to-probe interfaces with alternate options for both probe and probe instrument configurations, the personnel in this nucleus will be augmented as required by the activities. The special characteristics of the probe and probe instrument tasks in design interface activities and in integration test activities are discussed in Sections 3.2.2 and 4.2.2, respectively.

As a part of the organizational operating procedures, it is desirable to establish the methods for formal and informal communications between NASA/ARC, the experimenters, and the spacecraft and probe contractor with respect to experiment interface and integration activities. Figure A-4 shows the lines of communication in a schematic diagram. The NASA/ARC Pioneer project groups, represented by a series of lines at the left side of the figure, and boxes representing the spacecraft experimenters and the probe experimenters have been added to the contractor organization (Figure A-3), with lines drawn to show the channels of formal and informal communication. Significant relationships shown by the diagram are as follows: All experimenters (probe, probe bus, and orbiter) receive formal communication and direction only from NASA; contractor communication with experimenters will be initiated by NASA/ARC and confined to informal

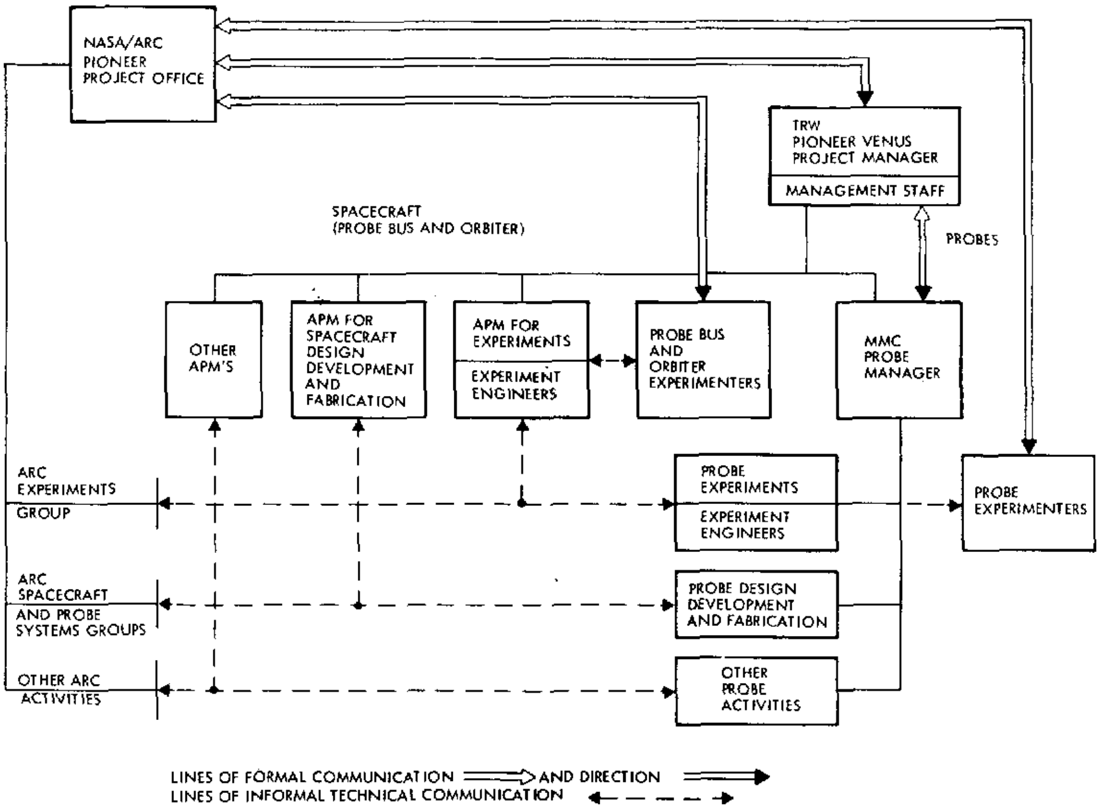


Figure A-4. Lines of Communication Between NASA/ARC, The Experimenters, and The Spacecraft and Probe Contractor for Experiment Interface and Integration Activities

technical discussion, exchanges of technical information, etc., with monitoring by, or report to, NASA/ARC as described further below. Similarly, formal communication by and with NASA/ARC regarding experiment interface and integration matters for the probes, probe bus, and orbiter, and formal communication between the TRW APM for experiments and probe personnel, including the experiment group, will be through the TRW project manager. To facilitate such communication the project manager's office will include a scientist/engineer staff function with specific responsibility for such communications. However, direct informal communication between NASA/ARC Pioneer project personnel and Martin Marietta probe personnel, as well as TRW spacecraft personnel will be encouraged, in experiment-related activities as in other Pioneer project activities. Such informal communications will most commonly be between the counterpart groups in the NASA/ARC and contractor organizations, as shown by the symmetrical layout of the diagram, with the focus for experiment interface and integration activities upon the experiment groups. There will be informal technical liaison with other groups when desirable.

Procedures for formal and informal communications are as follows:

- Formal communications will be written. Directions having the force of authority and responses thereto are formal communications. Submission and acceptance of contractually required documentation and evidence of contractually required activity performance are formal communications. Formal technical communications which do not have contractual impact will follow project office channels, while communications with contractual implications will be between corresponding Contracts personnel.

- Informal communications, on the other hand, consist most commonly of telephone calls and meetings of concerned personnel. In the case of telephone calls between contractor personnel and experimenter personnel (or their subcontractors) the former will notify the ARC Experiments Manager by telephone, with written confirmation as requested, of the fact and content of such communication. In the case of meetings between contractor and experimenter personnel, the former will notify the ARC Experiments Manager in advance, of the planned meeting (time, place, subject matter, and attendees), so that ARC representatives may attend if desired. After the meeting, if ARC representatives were not present, contracts personnel will notify the ARC Experiments Manager of the substance of the meeting.

2.3 Schedules

Preliminary schedules of interface, integration, and test activities are given for 1978 launches of both the multiprobe and orbiter missions, in Figures A-5 and A-6 and Tables A-1 and A-2. The figures show milestone schedules of the major events for the two missions, while the tables present a more detailed description of the various related activities involved in the milestone events, especially early interface definition activities which are not as specific in nature and time as the major milestone events.

These schedules are based upon the present Spacecraft Test and Integration Plan C, which provides for: 1) a structural/thermal model (STM) for the orbiter spacecraft, and similarly for the probe bus spacecraft; 2) an engineering test model (ETM) for each spacecraft; and 3) a flight spacecraft for each mission with an associated test program which serves the purposes of the customary prototype spacecraft testing combined with flight spacecraft testing; the flight spacecraft may be regarded as a proto/flight unit. According to this plan, STM, ETM, and flight model instruments are required for the probe bus and orbiter spacecraft experiments, and similarly, STM, ETM, and flight probes are required

for the probe mission, in which the complete probes are regarded essentially as additional instruments. However, the probe instrument development, test, and integration program employs the customary structural/thermal, prototype, and flight models, with fully qualified prototype probes (including instruments) preceding the flight probes. Thus, the probe instrument development, test, and integration program is phased somewhat in advance of the corresponding probe bus instrument activities. The time schedules given here will, of course, be modified as required when the spacecraft test and integration program is finally determined. Table A-3 shows the progression of instruments up to the flight unit and where they will be used on the spacecraft (bus or orbiter) and probes.

The activity schedules of Figure A-5 and Table A-1 for the probe mission are based upon a project go-ahead for the experiment implementation phase on 1 February 1974 and launch on 1 September 1978, where appropriate, the dates are specifically keyed to TRW Spacecraft Integration and Test Plan C.

The experiment definition phase at NASA will have been underway for up to 10 months prior to 1 February 1974. It is therefore assumed that preliminary experiment definition will be well advanced, and that the spacecraft and probe contractor will be supplied by NASA/ARC with a Preliminary Experiments Description Document and a Preliminary Spacecraft and Probe Requirements Document shortly after project go-ahead. As shown in Table A-1, the first contractor task to be performed is to update the baseline interface information which was contained in the study phase experiment interface documents and spacecraft and probe system specifications and to submit to NASA/ARC a Spacecraft and Probe—Scientific Instrument Interface Document. This update will be done as soon as possible in the first month after receipt of the aforementioned NASA/ARC documents in order to expedite early firming of the instrument-spacecraft/probe interfaces. The update will result in a Spacecraft and Probe—Scientific Instrument Interface Document written by TRW; to be of maximum use to ARC, it will be done as nearly as possible to the level of detail of a specification. Informal discussions of problem areas among ARC and contractor personnel is of great importance during the study and

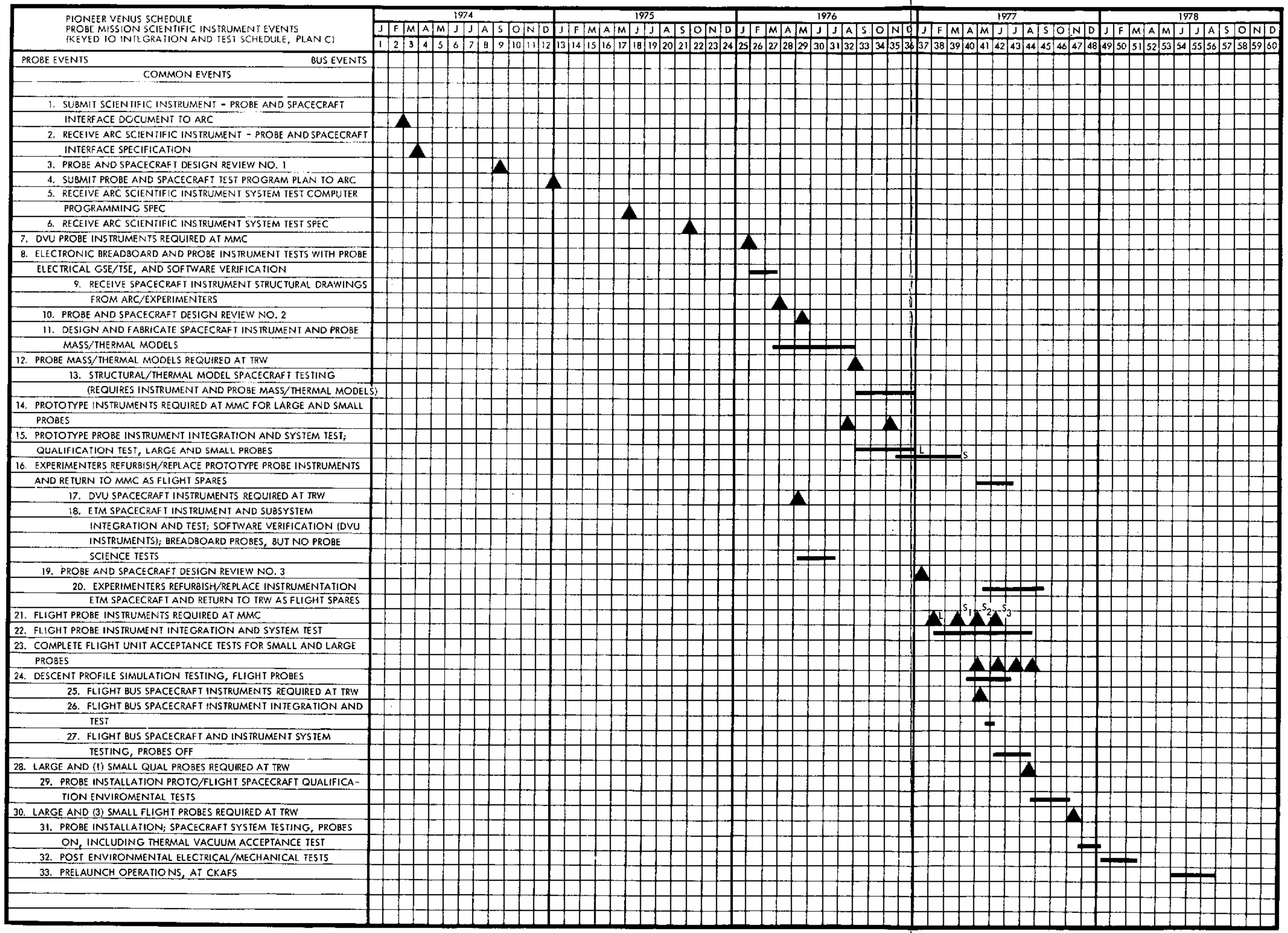


Figure A-5. Scientific Instrument Interfaces, Integration and Test Probe Mission Milestone Schedule of Major Events

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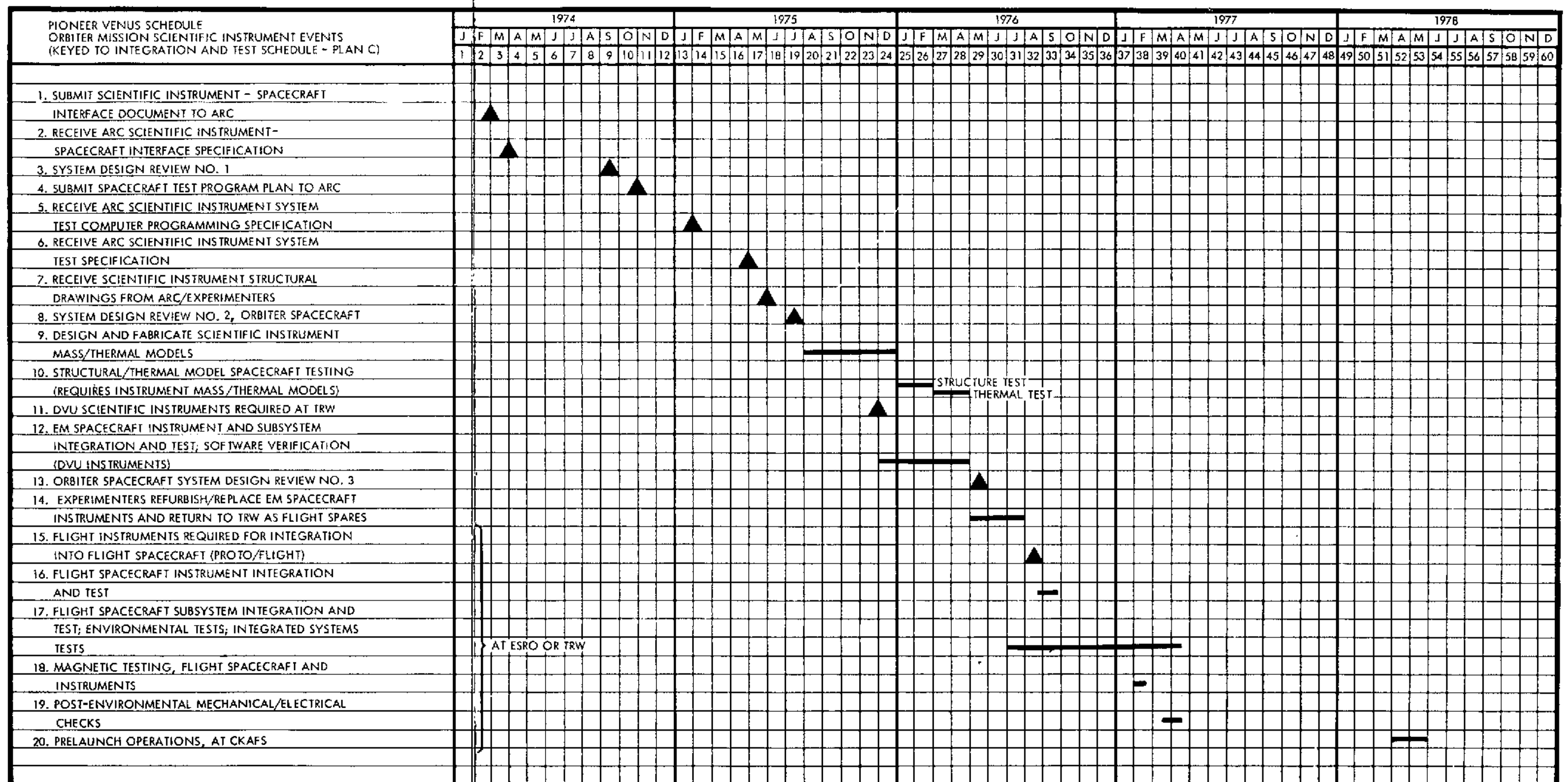

Figure A-6. Scientific Instrument Interfaces, Integration, and Test Orbiter Mission Milestone Schedule of Major Events

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Table A-1. Detailed Schedule of Scientific Instrument Interface, Integration and Test Activities and Correlated Spacecraft/Probe Activities — Probe Mission

EXPERIMENT EVENTS	SPACECRAFT AND PROBE EVENTS	DATE, BASED UPON GO-AHEAD 2/1/74	INTERFACE LIAISON NEEDED
RECEIVE ARC PRELIMINARY EXPERIMENTS DESCRIPTION DOCUMENT	RECEIVE ARC PRELIMINARY SPACECRAFT AND PROBE REQUIREMENTS DOCUMENT	ASAP (TENTATIVELY ASSUMED TO BE 2/4/74)	STUDY AND EVALUATION BY TRW/MMC EXPERIMENT GROUP IN COORDINATION WITH SYSTEM AND SUBSYSTEM ENGINEERS
	TRW SUBMIT SCIENTIFIC INSTRUMENT-PROBE AND SPACECRAFT INTERFACE DOCUMENT TO ARC	3/4/74	EVALUATION BY ARC, INFORMAL DISCUSSION AND COORDINATION WITH TRW/MMC
ARC SCIENCE EXPERIMENT MEETINGS AND REVIEWS		AS CALLED	ATTENDANCE/PARTICIPATION BY TRW/MMC EXPERIMENT MANAGERS AND ENGINEERS, AS REQUIRED BY ARC
	RECEIVE ARC SCIENTIFIC INSTRUMENT-PROBE AND SPACECRAFT INTERFACE SPECIFICATION	UPON COMPLETION OF ARC REVIEW OF SPACECRAFT AND PROBE INTERFACE DOCUMENT, ~4/1/74	
	TRW/MMC REVIEW INTERFACE SPECIFICATION AND SPACECRAFT AND PROBE DESIGN	BEGIN 4/74	DETAILED INFORMAL DISCUSSION AMONG TRW/MMC/ARC/EXPERIMENTER PERSONNEL*; IDENTIFICATION AND WORKING OF INTERFACE PROBLEMS, INCOMPATIBILITIES
EXPERIMENT SPLINTER MEETINGS AT TRW/MMC WITH EXPERIMENTER AND ARC REPRESENTATIVES		4/74-8/74, AS ARRANGED	
ARC SCIENCE EXPERIMENT MEETINGS AND REVIEW		AS CALLED	ATTENDANCE/PARTICIPATION BY TRW/MMC EXPERIMENT MANAGER AND ENGINEERS, AS REQUIRED BY ARC
	RECEIVE FROM ARC REVISIONS OF SCIENTIFIC INSTRUMENT-PROBE AND SPACECRAFT INTERFACE SPECIFICATION	1 TO 3 WEEKS IN ADVANCE OF PROBE AND SPACECRAFT DESIGN REVIEW NO. 1	COORDINATE WITH ARC IN PREPARATION FOR DESIGN REVIEW
	REVISE PROBE AND SPACECRAFT DESIGN SPECIFICATIONS, AS REQUIRED	FOR DESIGN REVIEW NO. 1	
	PROBE AND SPACECRAFT DESIGN REVIEW NO. 1	9/15/74	WORK RESULTING ACTION ITEMS
	SUBMIT PROBE AND SPACECRAFT TEST PROGRAM PLAN TO ARC	1/1/75	EVALUATION BY ARC, INFORMAL DISCUSSION AND COORDINATION WITH TRW/MMC
	RECEIVE ARC SCIENTIFIC INSTRUMENT SYSTEM TEST COMPUTER PROGRAMMING SPECIFICATION	6/1/75	TRW/MMC REVIEW SPECIFICATIONS AND ASSIST IN SOLUTION OF ANY INCOMPATIBILITIES IDENTIFIED
	RECEIVE ARC SCIENTIFIC INSTRUMENT SYSTEM TEST SPECIFICATIONS	10/1/75	

*EXPERIMENTER CONTACTS WHERE REQUIRED WILL BE INITIATED AND DIRECTED BY ARC

Table A-1. Detailed Schedule of Scientific Instrument Interface, Integration and Test Activities and Correlated Spacecraft/Probe Activities – Probe Mission (Continued)

EXPERIMENT EVENTS	SPACECRAFT AND PROBE EVENTS	DATE, BASED UPON GO-AHEAD 2/1/74	INTERFACE LIAISON NEEDED
DVU INSTRUMENTS FOR LARGE AND SMALL PROBES REQUIRED AT MMC	WRITE PROBE AND SPACECRAFT TEST PROCEDURES, AND WRITE AND DEBUG COMPUTER PROGRAMS, INCORPORATING INSTRUMENT REQUIREMENTS FROM ARC SPECIFICATIONS	6/75-7/76	COORDINATE WITH ARC/ EXPERIMENTERS*
	ELECTRONIC BREADBOARD AND PROBE INSTRUMENT TESTS WITH PROBE ELECTRICAL GSE/TSE, AND SOFTWARE VERIFICATION	2/2/76-3/26/76	
	COMPLETE PRELIMINARY DESIGNS OF PROBE SUBSYSTEMS: MECHANICAL ELECTRICAL	 3/1/76 4/16/76	DETAILED INFORMAL DISCUSSION AMONG MMC/TRW, ARC/ EXPERIMENTER PERSONNEL; IDENTIFICATION AND WORKING OF INTERFACE PROBLEMS, INCOMPATIBILITIES
	PROBE AND SPACECRAFT DESIGN REVIEW NO. 2	5/15/76	WORK RESULTING ACTION ITEMS
DESIGN AND FABRICATE MASS/THERMAL MODELS OF SPACECRAFT INSTRUMENTS AND PROBES		3/15/76-9/1/76	RECEIVE SPACECRAFT INSTRUMENT STRUCTURAL DRAWINGS AND POWER PARAMETERS FROM ARC/EXPERIMENTERS BY 4/1/76
	PROBE MASS/THERMAL MODELS REQUIRED AT TRW	9/1/76	
PROTOTYPE INSTRUMENTS FOR LARGE PROBE (LP) AND SMALL PROBES (SP) REQUIRED AT MMC		LP 8/13/76 SP 11/5/76	COORDINATE WITH ARC/ EXPERIMENTERS*
PROTOTYPE PROBE INSTRUMENT INTEGRATION AND SYSTEM TEST; QUALIFICATION TEST, LARGE AND SMALL PROBE		LP 9/1/76-1/1/77 SP 11/22/76-4/1/77	
EXPERIMENTERS REFURBISH/ REPLACE PROTOTYPE PROBE INSTRUMENTS AND RETURN TO MMC AS FLIGHT SPARES	REFURBISH PROTOTYPE PROBE COMPONENTS AS FLIGHT SPARES	5/1/77-7/22/77	INSTRUMENTS REFURBISHED/ REPLACED BY EXPERIMENTERS AS REQUIRED AND DIRECTED BY ARC, AFTER EVALUATION OF ALL TEST RESULTS
	STRUCTURAL MODEL SPACECRAFT TESTING	9/1/76-11/1/76	SPACECRAFT EXPERIMENT MASS/THERMAL MODELS AND PROBE MASS/THERMAL MODELS REQUIRED
	THERMAL MODEL SPACECRAFT TESTING	11/1/76-1/1/77	
DVU/ETM SPACECRAFT INSTRUMENTS REQUIRED AT TRW		5/1/76	COORDINATE WITH ARC/ EXPERIMENTERS.* REQUIRES EXPERIMENT TEST PROCEDURES AND SOFTWARE.
ETM SPACECRAFT INSTRUMENT AND SUBSYSTEM INTEGRATION AND TEST; SOFTWARE VERIFICATION WITH DVU INSTRUMENTS AND BREADBOARD PROBES		5/1/76-7/25/76	PARTICIPATION BY EXPERIMENT ENGINEERS AS INTEGRAL PART OF TEST CREW, UNDER DIRECTION OF TEST DIRECTOR, AND BY ARC EXPERIMENT PERSONNEL

*EXPERIMENTER CONTACTS WHERE REQUIRED WILL BE INITIATED AND DIRECTED BY ARC

Table A-1. Detailed Schedule of Scientific Instrument Interface, Integration and Test Activities and Correlated Spacecraft/Probe Activities – Probe Mission (Continued)

EXPERIMENT EVENTS	SPACECRAFT AND PROBE EVENTS	DATE, BASED UPON GO-AHEAD 2/1/74	INTERFACE LIAISON NEEDED
	PROBE AND SPACECRAFT DESIGN REVIEW NO. 3	1/15/77	WORK RESULTING ACTION ITEMS
EXPERIMENTER REFURBISH OR REPLACE INSTRUMENTS TESTED ON ETM SPACECRAFT AND RETURN TO TRW AS FLIGHT SPARES	REFURBISH ETM SPACECRAFT COMPONENTS AS FLIGHT SPARES	5/15/77-9/15/77	INSTRUMENTS RETURNED TO EXPERIMENTERS' FACILITIES AS REQUIRED AND DIRECTED BY ARC, AFTER EVALUATION OF ALL TEST RESULTS
FLIGHT PROBE INSTRUMENTS REQUIRED AT MMC	LARGE PROBE SMALL PROBE	2/18/77 3/25, 5/2, 6/3/77	COORDINATE WITH ARC/ EXPERIMENTERS*
FLIGHT PROBE INSTRUMENT AND SUBSYSTEM INTEGRATION AND SYSTEM TEST		2/18/77-8/26/77	
COMPLETE FLIGHT UNIT ACCEPTANCE TESTING			MONITOR INSTRUMENT PERFORMANCE DURING ENVIRONMENTAL TESTS AND EXERCISE INSTRUMENTS AS SCHEDULED IN COORDINATION WITH ARC/EXPERIMENTERS*
DESCENT PROFILE SIMULATION TESTING FOR FLIGHT PROBES		11/16/77-7/15/77	
FLIGHT SPACECRAFT INSTRUMENTS REQUIRED AT TRW		5/10/77	COORDINATE WITH ARC/ EXPERIMENTERS*
FLIGHT SPACECRAFT INSTRUMENT INTEGRATION AND TEST		5/20/77-6/3/77	PARTICIPATION BY EXPERIMENT ENGINEERS AS INTEGRAL PART OF TEST CREW, UNDER DIRECTION OF TEST DIRECTOR, AND BY ARC EXPERIMENT PERSONNEL AND P.I. REPRESENTATIVES, IN EXPERIMENT INTEGRATION AND TEST ACTIVITIES AT TRW, AND IN PRELAUNCH OPERATIONS AT CKAFS
	FLIGHT SPACECRAFT SUBSYSTEM AND SYSTEM TESTS, PROBES OFF	6/3/77-8/21/77	
	LARGE AND (1) SMALL QUAL PROBES REQUIRED AT TRW	8/1/77	
	PROBE INSTALLATION AND PROTO/FLIGHT SPACECRAFT QUALIFICATION ENVIRONMENTAL TESTS, PROBES ON	8/21/77-11/14/77	
	LARGE AND (3) SMALL FLIGHT PROBES REQUIRED AT TRW	11/14/77	
	PROBE INSTALLATION AND SPACECRAFT SUBSYSTEM AND SYSTEM TESTS, PROBES ON, INCLUDING THERMAL VACUUM ACCEPTANCE TESTS	11/11/77-1/15/78	
	PRESHIPPING MECHANICAL/ ELECTRICAL CHECKS, FLIGHT SPACECRAFT AND INSTRUMENTS	1/15/78-3/24/78	
	PRELAUNCH OPERATIONS	6/1/78-9/1/78	

*EXPERIMENTER CONTACTS WHERE REQUIRED WILL BE INITIATED AND DIRECTED BY ARC

Table A-2. Detailed Schedule of Scientific Instrument Interface, Integration and Test Activities and Correlated Spacecraft Activities — Orbiter Mission

EXPERIMENT EVENTS	SPACECRAFT EVENTS	DATE, BASED UPON GO AHEAD 2/1/74	INTERFACE LIAISON NEEDED
RECEIVE ARC PRELIMINARY EXPERIMENTS DESCRIPTION DOCUMENT	RECEIVE ARC PRELIMINARY SPACECRAFT REQUIREMENTS DOCUMENT	ASAP (TENTATIVELY ASSUMED TO BE 2/4/74)	REVIEW BY TRW EXPERIMENT GROUP IN COORDINATION WITH SPACECRAFT SYSTEM AND SUB-SYSTEM ENGINEERS AND INPUTS TO ARC
	TRW SUBMIT SCIENTIFIC INSTRUMENT-SPACECRAFT INTERFACE DOCUMENT TO ARC	3/4/74	EVALUATION BY ARC, INFORMAL DISCUSSION AND COORDINATION WITH TRW
ARC SCIENCE EXPERIMENT MEETINGS AND REVIEWS		AS CALLED	ATTENDANCE/PARTICIPATION BY TRW APM FOR EXPERIMENTS AND ENGINEERS, AS REQUIRED BY ARC
	RECEIVE ARC SCIENTIFIC DOCUMENT-SPACECRAFT INTERFACE SPECIFICATION	UPON COMPLETION OF ARC REVIEW OF SPACECRAFT INTERFACE DOCUMENT ~4/1/74	
EXPERIMENT SPLINTER MEETINGS AT TRW WITH EXPERIMENTER AND ARC REPRESENTATIVES	TRW REVIEW INTERFACE SPECIFICATION AND SPACECRAFT DESIGN	TWO MONTHS FOLLOWING RECEIPT OF INTERFACE SPECIFICATION	DETAILED INFORMAL DISCUSSION AMONG TRW, ARC, AND EXPERIMENTER PERSONNEL;* IDENTIFICATION AND WORKING OF INTERFACE PROBLEMS, INCOMPATIBILITIES
ARC SCIENCE EXPERIMENT MEETINGS AND REVIEWS		AS CALLED	ATTENDANCE/PARTICIPATION BY TRW APM FOR EXPERIMENTS AND ENGINEERS, AS REQUIRED BY ARC
	RECEIVE FROM ARC REVISIONS OF SCIENTIFIC INSTRUMENT-SPACECRAFT INTERFACE SPECIFICATION	1-3 WEEKS IN ADVANCE OF SPACECRAFT DESIGN REVIEW NO. 1	COORDINATE WITH ARC IN PREPARATION FOR DESIGN REVIEW
	TRW REVISE SPACECRAFT DESIGN SPECIFICATIONS, AS REQUIRED	FOR DESIGN REVIEW NO. 1	
	SPACECRAFT DESIGN REVIEW NO. 1	9/15/74	WORK RESULTING ACTION ITEMS
	SUBMIT SPACECRAFT TEST PROGRAM PLAN TO ARC	11/1/74	EVALUATION BY ARC, INFORMAL DISCUSSION AND COORDINATION WITH TRW
	RECEIVE ARC SCIENTIFIC INSTRUMENT SYSTEM TEST COMPUTER PROGRAMMING SPECIFICATION	2/1/75	TRW REVIEW SPECIFICATIONS AND ASSIST IN SOLUTION OF ANY INCOMPATIBILITIES IDENTIFIED
	RECEIVE ARC SCIENTIFIC INSTRUMENT SYSTEM TEST SPECIFICATIONS	5/1/75	
	WRITE SPACECRAFT TEST PROCEDURES, AND WRITE AND DEBUG COMPUTER PROGRAMS, INCORPORPORATING INSTRUMENT REQUIREMENTS FROM ARC SPECIFICATIONS		COORDINATE WITH ARC/EXPERIMENTERS*
	ORBITER SPACECRAFT SYSTEM DESIGN REVIEW NO. 2		WORK RESULTING ACTION ITEMS

*EXPERIMENTER CONTACTS WHERE REQUIRED WILL BE INITIATED AND DIRECTED BY ARC

Table A-2. Detailed Schedule of Scientific Instrument Interface, Integration and Test Activities and Correlated Spacecraft Activities — Orbiter Mission (Continued)

EXPERIMENT EVENTS	SPACECRAFT EVENTS	DATE, BASED UPON GO AHEAD 2/1/74	INTERFACE LIAISON NEEDED
TRW DESIGN AND FABRICATE INSTRUMENT MASS/ THERMAL MODELS FOR STRUCTURAL/THERMAL MODEL SPACECRAFT TESTING		8/1/75-1/1/76	RECEIVE INSTRUMENT STRUCTURAL DRAWINGS FROM EXPERIMENTERS BY 6/1/75
	STRUCTURAL MODEL TESTING	1/1/76-3/1/76	INSTRUMENT MASS/THERMAL MODELS REQUIRED
	THERMAL MODEL TESTING	3/1/76-5/1/76	
DVU/ETM SCIENTIFIC INSTRUMENTS REQUIRED AT TRW		12/1/75	COORDINATE WITH EXPERIMENTERS/ARC.* PARTICIPATION BY TRW EXPERIMENT ENGINEERS AS INTEGRAL PART OF TEST CREW, UNDER DIRECTION OF TEST DIRECTOR, AND BY ARC EXPERIMENT PERSONNEL. REQUIRES EXPERIMENT TEST PROCEDURES AND SOFTWARE.
ETM SPACECRAFT INSTRUMENT AND SUBSYSTEM INTEGRATION AND TEST; SOFTWARE VERIFICATION WITH DVU INSTRUMENTS		12/1/75-5/1/76	
EXPERIMENTERS REFURBISH/ REPLACE ETM SPACECRAFT INSTRUMENTS AND RETURN TO TRW AS FLIGHT SPARES	REFURBISH ETM SPACECRAFT EQUIPMENT UNITS AS FLIGHT SPARES	5/1/76-8/1/76	INSTRUMENTS RETURNED TO EXPERIMENTERS' FACILITIES AS REQUIRED AND DIRECTED BY ARC, AFTER EVALUATION OF TEST RESULTS
IF ESRO PARTICIPATES, TRW SHIPS TO ESRO FLIGHT SPARE INSTRUMENTS AND SPACECRAFT COMPONENTS		9/1/76	COORDINATE WITH ARC/ EXPERIMENTERS* AND ARC/ESRO
FLIGHT INSTRUMENTS REQUIRED AT TRW (OR ESRO)	FLIGHT SPACECRAFT (PROTO/ FLIGHT) REQUIRED (AT TRW OR ESRO)	8/15/76	
FLIGHT SPACECRAFT INSTRUMENT INTEGRATION AND TEST		8/27/76-9/24/76	PARTICIPATION BY TRW EXPERIMENT ENGINEERS AS INTEGRAL PART OF TEST CREW, UNDER DIRECTION OF TEST DIRECTOR, AND BY ARC EXPERIMENT PERSONNEL, IN EXPERIMENT INTEGRATION AND TEST ACTIVITIES AT ESRO (OR TRW), AND IN PRELAUNCH OPERATIONS AT CKAFS.
	FLIGHT SPACECRAFT SUBSYSTEM INTEGRATION AND TEST; ENVIRONMENTAL TESTS; INTEGRATED SYSTEMS TESTS	7/1/76-4/15/77	
	MAGNETIC TESTING, FLIGHT SPACECRAFT AND INSTRUMENTS	2/1/77-2/15/77	
	POST-ENVIRONMENTAL MECHANICAL/ELECTRICAL CHECKS	3/15/77-4/15/77	
	PRELAUNCH OPERATIONS	4/1/78-6/1/78	

*EXPERIMENTER CONTACTS WHERE REQUIRED WILL BE INITIATED AND DIRECTED BY ARC

Table A-3. Scientific Instrument Usage

INSTRUMENT	SPACECRAFT USAGE (BUS AND ORBITER)	PROBE USAGE
MASS/THERMAL MODELS*	STM	STM
DESIGN VERIFICATION UNIT	ETM, IF PROTOTYPE INSTRUMENT NOT AVAILABLE	ETM
PROTOTYPE	ETM	PROTO/QUAL
FLIGHT UNIT	PROTO/FLIGHT	FLIGHT

*SYSTEM-CONTRACTOR FABRICATED

in evaluation of the interface definition by ARC. Based upon such discussion, TRW will also prepare and submit a revised version of the Spacecraft and Probe—Scientific Instrument Interface Document as required by ARC. Upon completion of the review process, ARC will issue the Scientific Instrument—Probe and Spacecraft Interface Specification which, with subsequent revisions, remains the controlling instrument interface document throughout the remainder of the program. This milestone event will be scheduled by NASA/ARC; it is shown for purpose of illustration in the present preliminary schedule as occurring about 1 April 1974.

As stated in the Announcement of Flight Opportunity packet for Pioneer Venus experiments, Document PV-1005.00 issued 20 September 1972, NASA/ARC will hold two experiment project reviews per year, and other science experiment meetings at dates not yet determined. Because of the importance of early firming of instrument-spacecraft/probe interfaces and the contribution that such meetings can make to working interface problems, the first of such meetings has been arbitrarily shown in the schedule of Table A-1 before the issue of the interface specification. The second follows, and aids in the working of action items from, the first spacecraft and probe design review. Again, this schedule is for illustration only; ARC science experiment meetings and reviews and appropriate contractor participation will be as scheduled and required by ARC.

After issue by NASA/ARC of the Scientific Instrument—Probe and Spacecraft Interface Document, efforts to identify and resolve interface problems and incompatibilities will be continued through detailed, informal discussions among contractor, ARC, and experimenter personnel.

Experimenter contacts with the spacecraft system contractor will be initiated and directed where required by ARC. Under ARC direction, the contractor will hold a series of experimenter splinter meetings at TRW/ Martin Marietta, as described in Section 3, below. For greatest effectiveness they might best be held just before the first spacecraft design review, as shown in Table A-1; the scheduling is subject to change as developing needs indicate. In a similar fashion, probe experimenter splinter meetings will be held at Martin Marietta in Denver.

After the final spacecraft and probe design reviews, instrument interface problems will continue to be worked as necessary, but the emphasis shifts to pre-integration, integration, and test activities. The remaining items of Table A-1 consist of the instrument-related activities required by the spacecraft and probe integration and test program. These activities are highly specific in nature, and fall into a necessary and logical sequence of milestone events as shown b the tabulation of items below the third item in Figure A-5; in comparison, Table A-1 provides a description of the nature and method of accomplishment of the activities associated with each item in the milestone schedule chart of Figure A-5, with the inter-related requirements of experiment events and spacecraft/probe events indicated in detail.

The activity schedules of Figure A-6 and Table A-2 for the orbiter mission are similar to those of Figure A-5 and Table A-1 for the multiprobe mission, except for the absence of probe and probe instrument activities.

The remaining differences are those associated with the possible participation of ESRO with NASA in the orbiter mission, with the physical integration and system level testing of the scientific instruments and the flight spacecraft occurring at the ESRO facility in Holland. These appear in the latter part of Table A-2 and Figure A-6. There is no change in the ETM instrument and test activities, as these are carried out at TRW as before. Participation by: 1) TRW experiment engineers, who have worked with the experiment interface engineering since the beginning of the program, as an integral part of the test crew under the direction of the test director; and by 2) ARC experiment personnel in experiment integration and test activities at ESRO is especially important. They are needed to

support the ESRO integration and test personnel, who have not had previous experience with the probe mission integration and test activities.

For details of the overall integration and test program and its scheduling, refer to the TRW Spacecraft Integration and Test Plan; the aspects specifically pertaining to the scientific instruments are presented in Section 4.

3. SYSTEM AND DESIGN INTERFACE ACTIVITIES– PROBES, PROBE BUS, AND ORBITER

The purpose of these activities is to determine and evaluate the system requirements imposed by the experiments on the spacecraft/probe system design and to define the specific interfaces of the spacecraft and probes with each scientific instrument.

As indicated in Figure A-1, the scientific instrument design interface activities are the same, in concept, for spacecraft and for probe instruments. Hence, the description of activities in this section applies to spacecraft and probe instruments alike. These activities will begin promptly at go-ahead, assumed to coincide with the start of the experiment implementation phase by NASA/ARC.

The first contractor activity will be to update the experiment information from the Phase B study by a detailed review of the information defined by ARC during the experiment definition phase. The information that will be made available to the contractor is outlined in ARC Document PV-1005.00, Science Management Plan, 1977 Multiprobe Mission, Pioneer Venus. This information will include detailed preliminary experiment descriptions, definition of science instrument interface and support requirements, detailed preliminary design of science instrument and associated test equipment and software, and possible identification of long-lead instrument development requirements. Typically, this data may be compiled in a Preliminary Experiments Description Document and a Preliminary Spacecraft/Probe Requirements Document, or the equivalent, to be supplied to the spacecraft and probe contractor. Additional information may be obtained from NASA/ARC briefings, formal experiment meetings, and from selected experiment proposals and informal contact with experimenters.

The experiment and instrument descriptions will be reviewed to determine among other things the effect of the following on the spacecraft/probe system design:

- Probe and probe bus target requirements
- Required trajectories and orbits
- Attitude requirements and accuracies
- Data handling and sampling schedules
- Mechanical, electrical and thermal requirements
- Instrument operational requirements
- Preliminary instrument test and integration requirements.

Experiment requirements will be compared with the spacecraft and probe design capabilities to verify that interface requirements are met as stated at this time. This activity will be performed in close coordination with ARC to ensure that the requirements are met in the most cost-effective manner.

The results of this activity will be documented in the Spacecraft and Probe—Scientific Instrument Interface Document and submitted to ARC for review. After review, the document will be revised as required. ARC will then formally issue the Scientific Instrument—Spacecraft and Probe Interface Specification, which becomes thereafter, with revisions as issued, the basic control document for instrument interface design for the probes and for the spacecraft.

The contractor, using the appropriate system and subsystem engineers, will then carry out detailed review of these specifications and also the design of the spacecraft and probes. The purpose of this review will be to ensure that all instrument interfaces are adequately defined, that they satisfy the experiment requirements, and are compatible with probe and spacecraft capabilities. The results of the review will be presented at the science experiment section of the first formal contractor probe and spacecraft design reviews.

When design and interface incompatibilities are identified, a flexible combination of well established, effective procedures will be employed.

First there will be prompt, informal communication and discussion of problems and potential interface changes with ARC and, as directed, with experimenters and their engineering staffs in coordination with ARC. Secondly, interface changes under control of NASA/ARC will be considered. In the latter case, the following procedure will be employed.

After the first official issue by ARC of the Scientific Instrument—Probe and Spacecraft Interface Specification, all changes thereto will be issued by ARC after evaluation of formal change requests. Such requests may be initiated by experimenters/ARC on the one hand, or by the contractor on the other. Based upon the proven simple and effective Pioneer 10 and 11 procedure for such requests, if initiated by TRW as spacecraft and probe contractor, an engineering change proposal form will be submitted to ARC; this describes the requested change and includes an estimated dollar cost and associated required action date, in sufficient detail for evaluation by ARC and coordination/discussion with the experimenter(s) affected. If the change request is initiated by ARC/experimenter(s), a proposed change of project documentation form will be submitted by ARC to TRW for evaluation in terms of spacecraft and/or probe technical and cost impact. Response will be by formal letter from the TRW Pioneer Venus project office. In either case, evaluation of the proposed change by all parties concerned may involve informal discussions and analyses by experimenters, ARC, and contractor personnel. If approved by ARC, the change is implemented by the issue by ARC of a revision to the Scientific Instrument—Probe and Spacecraft Interface Specification; in most cases this is efficiently accomplished by sending single replacement pages with a marginal number notation to indicate the changed section(s) and suitable date and designation data in standard form as they appear at the bottom of each page. The contractor will be responsible for transmitting and implementing all revision changes within the spacecraft and probe system and subsystem design and engineering groups. As formal documents, engineering change proposals, proposed change of project documentation forms, and specification revisions approved by ARC will be transmitted through the TRW Pioneer Venus project office.

Preliminary scientific instrument integration and test information will be a part of the instrument interface and requirements data received

and worked, but the emphasis initially will be on the spacecraft/probe design and interfaces, and will gradually shift to the instrument integration and test program as the project advances, as shown by the detailed schedules given in Section 2.3, above.

Scientific instrument interface management requires participation by contractor personnel in the following science experiment reviews and meetings:

1) <u>ARC Project Reviews</u> — two meetings at ARC during the experiment definition phase are listed in the Announcement of Flight Opportunities for the Pioneer Venus Multiprobe Mission. It is assumed that similar meetings will be held during the experiment implementation phase, and similarly for the orbiter mission. Attendance by the TRW APM for experiments, the MMC probe experiment manager, and staff engineers as required.

2) <u>ARC Science Experiment Meetings (both informal working meetings and formal design reviews)</u> — attendance by the TRW/MMC experiment managers, as appropriate for the subject experiments, with corresponding staff engineers.

3) <u>Preliminary and Final Spacecraft and Probe Design Reviews Held at TRW and MMC</u> — TRW procedure is essentially that employed successfully on the Pioneer 10 and 11 program and for details refer to the reliability program described in Volume II of this study. It provides for two preliminary reviews (a conceptual and a research and development review) and a final review, with the possibility of combining Nos. 2) and 3) for some areas, such as MGSE. Attendance and monitoring by ARC personnel is invited and expected; personnel from the TRW/MMC experiment groups will attend these reviews.

In addition to the above reviews and meetings, TRW under ARC direction will hold a series of experiment splinter meetings at TRW, as required, for several experimenters and/or their representatives and engineers (and possibly subcontractor engineers), with ARC representatives present to monitor and participate. Previous experience strongly indicates that some interface design problems can be worked most effectively and economically when spacecraft system and subsystem specialists are available as needed for each individual problem, and the entire gamut of problems relating to each experiment can be worked in detail at one time. It is most helpful to hold such meetings well in advance of formal design reviews in order to reduce greatly the number of unresolved problems and action items resulting from the formal review. The first

of such meetings might well be held immediately after the intensive review by TRW of the spacecraft design and the Spacecraft—Scientific Instrument Interface Specifications and before the first spacecraft design review. In the case of identified special problems, meetings for specific experiments may profitably be held prior to or following NASA/ARC science experiment meetings, as requested by TRW and arranged by ARC.

Although the splinter meetings will be informal working meetings, they will be planned carefully for a few experiments at a time, with attendees and agendas circulated in advance, and with the sequence of such meetings extending over several weeks. Similar splinter meetings for probe experiments will be held at MMC.

It is probable that experimenters or their contractors will hold the equivalent of preliminary design reviews on each instrument, as well as breadboard demonstrations. It is desirable that representatives of the spacecraft and probe contractor's experiment interface and integration group participate to an extent determined by NASA/ARC.

4. INTEGRATION AND TEST ACTIVITIES

In a narrow sense, scientific instrument integration refers to the physical mating of the instruments with their corresponding spacecraft or probe vehicles. In a broader sense, this term also includes all the necessary pre-integration activities which prepare for and lead to a successful integration and test sequence, together with associated activities such as test evaluation, instrument calibration, determination of baseline performance parameters, etc. Pre-integration and test activities (including definition of instrument GSE requirements by ARC/experimenters, fabrication of instrument GSE, test planning, and software preparation and verification for experiment GSE) involve extensive interface liaison activities between ARC/experimenters and contractor personnel and are initiated early in the program. Similarly, early testing with the structural/thermal model probes and spacecraft, fabrication and testing of prototype probes and probe instruments and the ETM spacecraft and instruments are closely related with, and flow from, continued interface liaison work directed toward integration and test activities.

Experiment engineers who have worked in the design interface activities from the start of the program will become experiment test engineers when the scientific instruments are delivered to TRW/MMC. This takes full advantage of the experience and knowledge gained during the early phases of the project and provides continuity throughout all instrument-related activities. It is anticipated that NASA/ARC will provide experiment test engineers to support the experiment test activities of the contractors. The number of contractor experiment test engineers required will, of course, depend on the size of the NASA/ARC support group.

Upon reaching the actual spacecraft integration and test phase of the program, the integration and test activities will be under control of the APM for integration, test, and launch, with the TRW/MMC experiment groups supplying a support function for information and aid in working instrument-related problems and maintaining the necessary instrument support facilities and documentation.

4.1 Probe Bus and Orbiter

The scientific instrument integration and test activities for the probe bus and orbiter spacecraft are part of the overall system integration and test program for each spacecraft. These programs are described in detail in the TRW Spacecraft Integration and Test Plan. The GFE flight instruments are delivered to TRW (or ESRO, in the case of European participation in the orbiter mission); ETM instruments are delivered to TRW for both missions. In the case of the probe mission spacecraft, the probes are delivered to TRW by MMC, with the scientific instruments integrated and tested as described in Section 4.2 below; the probes are regarded essentially as spacecraft system units, as described in the above-referenced Spacecraft Integration and Test Plan.

TRW will provide office space for ARC/experimenter personnel convenient to the integration area and an experiment laboratory will be available for experiment bench tests, auxiliary equipment storage, etc. Experimenters or their representatives and ARC representatives will participate in all bench tests, experiment integration and compatibility verification, and in all special experiment checks. ARC/experimenter

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representatives will review data at TRW during or subsequent to the performance of all systems verification operations. The experiment integration engineers will maintain a log of all experiment operations during the integration and test program. Similar procedures will be followed to the fullest extent possible if the integration of the orbiter flight spacecraft occurs at the ESRO facility, as will be arranged by detailed liaison with ARC/ESRO. The prelaunch test program for scientific instruments to be carried out at ETR will again be an integral part of the complete integrated system test program at the launch site and will include selected parts of the scientific instrument test program outlined here.

Following is an outline of that part of the complete spacecraft system integration and test program which is concerned specifically with the probe bus and orbiter scientific instruments.

TASK 1: RECEIVING

The scientific instrument is received by the NASA/experiment test engineer (ETE) in the Experiment Laboratory. Integration planning and property are notified and an R&O is generated specifying receiving-inspection, mass properties determinations and, in the case of DVU and prototype instruments, RFI testing is required.

TASK 2: INSPECTION

Quality Assurance and mass properties are notified. Visual inspection of the instrument is performed and discrepancies noted on the R&O. Weight and center of gravity determinations are made and data recorded on the R&O. The instrument is then submitted for RFI testing as required.

TASK 3: SIMULATOR CHECK

Upon successful completion of Tasks 1 and 2, the instrument is returned to the Experiment Laboratory and a simulator check is performed. The instrument is held in the Experiment Laboratory until the schedules require it for integration on the spacecraft.

TASK 4: INTEGRATION ON THE SPACECRAFT

The instrument is delivered to the spacecraft integration area by the ETE and integration performed per formal NASA-approved procedures. The integration of the instrument consists of a number of tests as shown below:

1) Mechanical Interface. When the instrument is placed on the spacecraft for monitoring, the footprint is verified, mechanical interference with respect to adjacent equipment, aperture opening, etc., are checked. Torquing of mounting hardware is performed per the platform installation drawings.

2) Bonding Resistance. Using a low range resistance meter, the bonding resistance between the interface connectors and spacecraft ground is measured, recorded, and verified to specification limits.

3) Fault Voltage—Spacecraft Side. A series fuse box is installed between the instrument and the spacecraft interface. All lines between the instrument and spacecraft are opened. Spacecraft power is turned on and fault voltage is measured using a DVM at each interface signal line at spacecraft side and verified within specification limits.

4) Fault Voltage—Instrument Side. The primary power and return lines on the ITC are closed to permit instrument power turn on. A series fuse box, fuse equivalent to the fuse requirement for the particular instrument, is switched into the power line and instrument power commanded on. Fault voltages are then checked as for the spacecraft side described in 3) above.

5) Instrument Power Profile

   a) Normal Turn-On Mode. With the fault voltages verified satisfactory, the power profile of the instrument is then checked. On the SFB, the primary power line is opened and a 0.1-ohm resistor installed in series across it. A 0.1-ohm resistor is used to minimize voltage drop. A DVM is placed across the resistor to measure the voltage drop, which is then converted to current. All other signal lines to the instruments are closed on the SFB so the instrument is totally functional. The spacecraft and instrument power are turned on and current into the instrument verified as nominal, whereupon the SFB fuse is switched out of the circuit to eliminate the voltage drop across it. Current and voltage then are recorded and the instrument input power calculated.

b) Other Instrument Modes. Other instrument operating modes which increase instrument power consumption are exercised and power consumption calculated as in a) above. Instrument and spacecraft power are turned off in preparation for the next test.

c) Instrument Turn-On. The test configuration is maintained and instrument power is turned on. The in-rush is photographed and, from the photographs, in-rush current is verified within specification limits. Instrument and spacecraft power are turned off in preparation for the next test.

6) Inrush Current

a) Spacecraft Power Turn-On. Since the primary bus power is supplied to the instrument when spacecraft power is on, it is desirable to check the in-rush surge current when spacecraft power is applied to the instrument inertia circuitry. Excessive in-rush currents can degrade the solid state fuses used on the spacecraft to fuse the instrument power bus.

The 0.1-ohm resistor is replaced with 1.0 ohm to increase measurement sensitivity and accuracy. Oscilloscope leads are placed across the resistor using 1X probes. Spacecraft power is turned on and the in-rush current photographed. Spacecraft in-rush current is measured from the photograph. The photograph becomes part of the test data package. Spacecraft power is left on in preparation for the next test.

7) Interface Signals. The tests are designed to verify both instrument and spacecraft interface signal parameters with both connected (under load). The SFB is removed and an interface test connector, cable, and breakout box installed to provide access to each individual signal line. Spacecraft power is turned on and the signal lines are checked as follows:

a) Power On/Off Line. The steady state value of this signal is checked with instrument power off using a DVM. Instrument power is turned on and the on-state value is recorded.

b) Analog Signals. All analog signals are then checked to verify operation. Values obtained under ambient test conditions are recorded.

c) Pulse or Digital Signals. Using an oscilloscope with 10X probes, all pulse signals are measured to verify proper limits of rise time, fall time, amplitude, and duration.

d) Pulse Commands. Using an oscilloscope, proper command characteristics are verified for pulse commands regarding rise and fall time, amplitude, and duration. Functional verification of commands is not obtained at this time since software has not been verified.

8) Noise Tests. Spacecraft or instrument generated noise transmitted on or coupled through to interface lines is measured for each instrument in turn with all other instruments disconnected. This test is performed in this way to provide a noise baseline since it has been found that as each instrument is connected, noise characteristics will vary.

   Using an oscilloscope, the peak-to-peak noise on each interface line is recorded. Where specification limits are exceeded, a photograph is taken and the amplitude and repetition rate of the noise recorded.

   A final noise measurement test is performed in the test described in the next section.

9) Instrument Data Check. Instrument and spacecraft power is turned off. The ITC's and breakout box are removed and the spacecraft harness mated to the instrument. Spacecraft and instrument power are turned on. The spacecraft and instrument are configured per specification requirements and a data check performed to verify proper operation of the instrument.

   In the engineering and prototype phase of the program, this test will also be used to validate the computer programming requirements for each instrument.

TASK 5: DETAILED INSTRUMENT TESTS

These tests, designed to verify instrument operation under most spacecraft and instrument operating modes, are detailed below:

- Spacecraft turn-on and configuration to nominal flight configuration.

- Instrument turn-on, data check, and calibration with and without stimulus, as required.

- Instrument-to-instrument interference test wherein the instrument under test is configured to a preferred mode and all other instruments are exercised through power off/on and all operational modes to determine any interference between instruments.

- Spacecraft-to-instrument interference wherein again the instrument under test is placed in a preferred mode. The spacecraft subsystems are then operated throughout various modes as in flight to determine any interference with the instruments.

- "Special instrument test requirements" is a section of the test designed to provide additional testing unique for particular instruments, such as special calibration of detectors, of bit rate, format tests, etc.

TASK 6: INTEGRATED SYSTEMS TEST NO. 1

The integrated system test is a combined spacecraft/instrument test and is used throughout the spacecraft test program to provide quick-look reference data points to establish any instrument degradation throughout the program as well as subsequent to major systems test. The test itself is divided into two sections, engineering and science:

- The engineering test simulates a mission profile of launch, orientation, and flight, and verifies preparation of the spacecraft subsystems.
- The science test simulates a flight profile. Instruments are exercised and stimulated with GSE, radioactive, or on-board instrument stimulus to simulate flight data. It is recommended that the use of hardline GSE stimulus be minimized so as to maintain a simulated flight and simplify test operations. GSE has proven to be a fairly unreliable test tool on past spacecraft programs.

TASK 7: INSTRUMENT ALIGNMENTS

Those instruments having optical alignment requirements are aligned to specification requirements using a spacecraft alignment facility designed and developed for this purpose. Alignments are performed at this time to verify integrity during the subsequent vibration test.

TASK 8: SPACECRAFT VIBRATION TEST

The vibration test is performed to verify spacecraft survival of the simulated launch environment.

The spacecraft is transferred to the vibration facility where it is configured for vibration testing in accordance with launch vehicle requirements.

The test is performed once in each of the three spacecraft axes. The spacecraft power is on and configured to a typical launch configuration. Spacecraft data is monitored throughout the vibration period. Instruments are not normally on during launch and therefore data is not monitored. A functional check of both spacecraft and instruments is normally performed after vibration in each spacecraft axis. These tests are abbreviated functional checks. A more detailed look at instruments is performed during the post vibration IST (See Task 10).

TASK 9: ALIGNMENT VERIFICATION

This test is performed to verify that alignment requirements are still within specification limits subsequent to vibration testing.

TASK 10: INTEGRATED SYSTEMS TEST NO. 2

(See Task 6 for detailed description of IST tests).

TASK 11: SPACECRAFT MAGNETICS TESTING

The magnetic tests are a series of tests designed to place the spacecraft in a magnetically clean state and measure the residual magnetism of the spacecraft. These tests consist of the following:

- Spacecraft functional test performed to verify spacecraft operation subsequent to shipment to the magnetic test site
- Spacecraft operating modes
- Spacecraft magnetized test
- Spacecraft demagnetized test

TASK 12: INTEGRATED SYSTEMS TEST NO. 3

This IST is performed to provide baseline data prior to thermal vacuum testing. (See Task 6).

TASK 13: THERMAL VACUUM TESTING

This test is performed to verify operation of the spacecraft system under simulated space environment in addition to verifying the thermal design.

The use of instrument GSE for this test is not recommended due to the complexity of the test configuration. The use of on-board calibration or stimulus sources is recommended.

TASK 14: INTEGRATED SYSTEMS TEST NO. 4

This IST is performed to provide data to evaluate effects of the thermal vacuum environment on the spacecraft and scientific instruments (see Task 6).

TASK 15: REMOVE SCIENTIFIC INSTRUMENTS FOR CALIBRATION

Subsequent to thermal vacuum tests, the instruments may be removed for detailed calibration by the experimenter.

TASK 16: REINTEGRATE SCIENTIFIC INSTRUMENTS

Upon return of the instruments to TRW, numerous tasks will be repeated such as Task 1: Receiving, Task 2; Inspection, Task 3: Simulator Check, and Task 4: Integration on the Spacecraft.

TASK 17: POST-ENVIRONMENTAL AND ETR INSTRUMENT TESTS

These tests are performed to provide instrument baseline operational data for comparison subsequent to shipment of the spacecraft to ETR. They are similar to the detailed instrument tests described in Task 5 with the deletion of the interference portions of those tests.

TASK 18: INTEGRATED SYSTEMS TEST NO. 5

An IST is performed at this time to verify overall systems operation and provide baseline data for the IST to be performed at ETR (see Task 6).

TASK 19: FINAL ALIGNMENT OF SCIENTIFIC INSTRUMENTS

Since some or all of the scientific instruments may have been removed (Task 15), realignment will be required.

Based upon TRW's previous experience, a list of recommendations for the Pioneer Venus program to minimize scientific instrument problems in the spacecraft integration and test program is given in the following paragraphs.

INSTRUMENT DESIGN

1) <u>Mechanical — General</u>

a) Consideration should be given to the use of uniform mounting hardware for instruments. Use of nonstandard hardware such as long feedthrough or specially machined hardware should be discouraged or prohibited.

b) Standard mounting tab configuration and size should be adopted.

c) Mounting tab location with respect to spacecraft platform location should be considered regarding access to mounting hardware for ease of installation and removal.

d) Aperture or telescope covers should be designed for easy-off, easy-on handling, and should consider multiple function design, such as protective cover and source holder for test operations.

e) Purge tube design should consider environmental constraints. Purge tubes should be incorporated into aperture or telescope design wherever possible, otherwise access, when in the spacecraft compartment, should be a consideration.

f) Access to mounting hardware and all interface harness connectors must be considered. All harness connectors must be at least 1-1/2 inches above platform. If access to test connectors is required during systems tests, these connectors must be mounted on top of instrument package. These accessibility requirements must be provided, and other requirements must accommodate to this.

g) Mounting of physical stimuli (calibration sources, light stimuli, etc.) must consider location of, and sensors' penetration of, the spacecraft wall, and must consider adjacent black boxes and their test requirements; e.g., the size of adjacent test consoles must be considered.

2) <u>Mechanical — Alignments</u>

a) Instrument design should incorporate means of achieving alignment requirements without the use of shims, to avoid affecting platform bonding resistance.

b) The alignment parameters should be expressed in terms identical to the spacecraft coordinate system orientation, and realistic alignment requirements must be defined accurately and early.

c) Consideration should be given to incorporating alignment targets into the design of the instrument, such as combining target mirrors on telescope covers or providing bench marks on instruments without telescopes.

3) Electrical—General

a) Instrument power turn on circuitry should be designed similar to the Pioneers 10 and 11, viz. 28 VDC is supplied to the instrument when spacecraft is turned on. Instrument turn on/off accomplished in the return side, by controlling 28 VDC to clock oscillator in instrument converter and down stream from filter networks to avoid large in-rush problems.

b) Instrument input voltage should be specified at the instrument connector, or at other clearly specified location.

c) Instrument to spacecraft interface circuitry must be provided for each scientific instrument on both the instrument and spacecraft sides of the interfaces.

d) Input and output impedances with specified minimum/maximum tolerances must be provided for each interface circuit for each instrument.

e) Instrument power requirements with minimum/maximum tolerance must be specified and tightly controlled. Periodic updates to specified power requirements should be provided to the spacecraft contractor. Consideration should be given to the desirability of specifying instrument power as watts or as current with a nominal 28 VDC input, but with minimum/maximum tolerance in either case.

f) Realistic fault voltage data must be provided to eliminate misinterpretation and confusion.

g) Rise and fall times for all digital circuits must be specified precisely and realistically at the instrument interface at a prescribed input and output impedance and method of measurement (particularly important for word gate fall times).

h) Instruments should be designed with appropriate reset circuitry to configure to a preferred status upon instrument power turn-on.

i) Spacecraft functions must never be directly connected to instrument test connectors without isolation provision to prevent a dead short which would clobber the entire spacecraft.

4) Connectors

a) General

- Wherever possible, the use of standard connectors should be established.

- Use of end pins should be discouraged to reduce the possibility of insulator block cracking of female connectors.

- The use of keyed connectors should be considered, e.g., the D-series connectors can be keyed by the use of blank pins in selected locations.

b) Interface Connectors

- Access to the instrument should be considered when locating interface connectors.

- If more than one connector is used, sufficient space should be provided between connectors to allow use of a demating tool.

- Interface connectors should be located at least 1-1/2 inches above the mounting platform.

- The use of connector savers in the form of an ITC should be considered to minimize mating-demating on the instrument side and permit test point access if required.

c) Test Connectors

- Use of common power test connector with associated in-flight jumper (for current monitor) and fuse connector should be considered. Fuse connector must be located so as to be accessible for current measurement. Access to these connectors should be provided from outside the spacecraft, preferably at the main test connector panel.

- Test connectors should be located to provide easy access for GSE cables or test plugs.

TEST EQUIPMENT DESIGN

1) GSE

   a) Design of GSE which must provide a stimulus in close proximity to the spacecraft must consider space available in the area of the spacecraft when in test configuration. Consideration should be given to the requirement of similar equipment in adjacent areas.

   b) Stimulus equipment as described above should be provided with suitable stands or tripods for all test configurations.

   c) If the GSE requires synchronizing or clocking pulses from the spacecraft or system test set, use the same terminology to avoid confusion.

2) Peripheral Equipment. Peripheral equipment such as scopes on signal generators should be designed in or supplied separately to avoid possible test delay should such equipment not be available by the contractor at time of test.

TEST ENVIRONMENT

1) Temperature

   a) Normal Test Environment. Temperature limits of the normal test environment should be established early to allow planning for any special requirements.

   b) Thermal Vacuum Test Environment. Temperature limits during thermal vacuum testing should be provided early to permit incorporation into test planning operations and procedures.

2) Humidity. The affects of humidity on instrument sensors should be established prior to start of the formal test program to permit proper planning for control, or design and fabrication of special fixtures or equipment to maintain the requirements as established.

3) Chemicals. Instrument susceptibility to chemical cleaning agents, propellant or other agents must be identified early in the program to prevent inadvertent damage.

4) Spacecraft Configuration. Test requirements for individual instruments should consider normal test configuration of the spacecraft. Instrument tests that require the spacecraft to be placed in an abnormal position should be discouraged due to the possibility of damage during handling, etc.

4.2 Probes

The tasks associated with the integration and test of probe instruments with the probes are basically similar to those discussed in Section 4.1 for probe bus and orbiter instruments. The differences in scheduling in order to allow fully integrated probes to be delivered for ETM and flight spacecraft integration and test have been outlined in Section 2.3, and shown in the schedules of Figure A-5 and Table A-1. In addition, there are differences in the interrelationships among prototype, qualification, and acceptance testing and the utilization of these test results; these differences are shown in the task summaries given below.

The experiment integration group under its experiment manager will continue at MMC into and through the integration and test phase of the program. The integration of the scientific instruments into the probes will be performed at MMC in Denver, and the GFE instruments should be physically delivered directly to MMC. The size of the group during this phase will be governed to some extent by the number of NASA/ARC experiment engineers in residence at MMC. During the spacecraft integration activities, a small number of MMC experiment engineers will be present at TRW as needed during appropriate portions of the all-up (i.e., with probes installed) portions of the probe bus ETM and flight system integration test programs.

The first group of tasks consists of those general activities associated with pre-flight-model instruments. These include pre-integration testing and may have an impact upon the latter stages of design activities. They are:

ACTIVITY NO. 1 AND 2: DESIGN AND PRODUCE EXPERIMENT BENCH TEST HARDWARE AND SOFTWARE; DESIGN PROBE ELECTRONIC GSE/TSE

These two tasks are scheduled to be completed in the third quarter after go-ahead. Liaison between the experimenter (and his contractor) and the probe system contractor is highly desirable in this effort to minimize duplication and maximize utilization of developed hardware and software.

ACTIVITY NO. 3: CONDUCT PROBE ELECTRICAL TEST MODEL (ETM) TESTS

Testing of the probe ETM is currently scheduled to begin in the 16th month after go-ahead. Qualification tests of the experiment engineering models (design verification unit, DVU) is scheduled for completion by the end of the 12th month after go-ahead. The present plan is to utilize the DVU of each experiment in the probe ETM. Therefore, these units, along with one set of GSE, must be made available immediately after completion of their qualification tests. To rapidly integrate these instruments into the ETM and ensure their proper functioning, it will be necessary to have personnel present who are familiar with each instrument. These can either be furnished by the experimenter, by NASA, or by the system contractor. In any event, liaison between personnel familiar with the ETM and instruments will be required.

ACTIVITY NO. 4: INFORMATION FOR THE FABRICATION OF PROTOTYPE INSTRUMENTS

Since the ETM tests are a verification of the interface between the probe subsystems and the instruments, the results of the test should be made known to the experimenter in a form and in sufficient time for him to consider the results with fabrication of his prototype unit. The results may require some change to the instrument or probe interfaces. For this purpose we would prepare, and furnish to NASA for distribution to each experimenter, a copy of the ETM test results pertinent to his instrument.

ACTIVITY NO. 5: PROTOTYPE INSTRUMENT INTEGRATION AND SYSTEMS TESTS

This experiment event correlates directly with the qualification test, large probe and small probe. These events occur 31 months after go-ahead for the large probe and approximately 3 months later for the small probe. The instruments should be delivered to the probe contractor 2 or 3 weeks prior to the system qualification test to allow for functional checkout tests, main science equipment assembly tests, and physical integration with the probe. This work will be supervised by personnel from MMC's experiment integration group, who have by this time become quite familiar with the operation of the instrument. Leading up to this activity, the contractor will have obtained instrument test specifications from NASA/ARC. These procedures will be integrated into the system level qualification test procedures. Liaison effort with the experimenter and his test engineer may be needed to develop specialized techniques for checking out the instrument in the system configuration where the bench test points will no longer be available.

The second group of tasks consists of the integration and test activities for the flight instruments with the large and small flight probes.

This phase begins with the receipt of the flight instruments nominally 36 months after go-ahead (several months later for the small probes, as shown in the milestone schedule). The tasks and procedures of this phase are not entirely new, as much of the same work was done for the earlier versions of the science instruments in the preceding phase. The following is a brief but detailed breakdown of the major tasks pertaining to the science instruments. It is not a complete list or description of all tests and omits some of the repetitive sequences conducted at the system level.

TASK 1: RECEIVING INSPECTION

The scientific instrument is received by the experiment test engineer in the Experiment Laboratory. Quality assurance and mass properties are notified. Visual inspection of the instrument is performed and discrepancies noted. Weight and center of gravity determinations are made and data recorded. The instrument is then submitted for RFI testing, if required.

TASK 2: SIMULATOR CHECK

Upon successful completion of Task 1, the instrument is returned to the Experiment Laboratory and a simulator check is performed. The instrument is held in the laboratory until required for integration into the main science equipment assembly.

TASK 3: MAIN SCIENCE EQUIPMENT ASSEMBLY SUBSYSTEM COMPATIBILITY TESTS

Science instrument electronics (in some cases without the sensors) will be assembled to the probe equipment shelf. Instruments having separated sensors such as the shock layer radiometer or temperature gauges shall have these sensors connected via adapter cables or use sensor simulators. For some of these tests the main science equipment assembly will be integrated with the communication and power equipment assembly.

1) Mechanical Interface. When the instrument is placed on the shelf for monitoring, the footprint is verified, and mechanical interference with respect to adjacent instruments is checked. Torqueing of mounting hardware is performed per the installation drawings.

2) Bonding Resistance. Using a low range resistance meter, the bonding resistance between the interface connectors and probe ground is measured, recorded, and verified to specification limits.

3) Fault Voltage – Probe Side. A series fuse box is installed between the instrument and the probe interface. All lines between the instrument and probe are opened. Probe power is turned on and fault voltage is measured using a digital volt meter at each interface signal line at the probe side and verified within specification limits.

4) Fault Voltage – Instrument Side. The primary power and return lines on the series fuse box are closed to permit instrument power turn on. A series fuse box fuse equivalent to the fuse requirement for the particular instrument is switched into the power line and instrument power commanded on. Fault voltages are then checked as for the probe side as described in 3) above.

5) Instrument Power Profile

a) Normal Turn-On Mode. With the fault voltages verified satisfactory, the power profile of the instrument is then checked. On the series fuse box the primary power line is opened and a 0.1-ohm resistor installed in series across it. A 0.1-ohm resistor is used to minimize voltage drop. A digital volt meter, used to measure voltage drop across the resistor, which can be converted to current is placed across the resistor. All other signal lines to the instruments are closed on the series fuse box so the instrument is totally functional. The probe and instrument power are turned on and current into the instrument verified as nominal whereupon the series fuse box fuse is switched out of the circuit to eliminate voltage drop across it. Current and voltage then are recorded and the instrument input power calculated.

b) Other Instrument Modes. Other instrument operating modes which increase instrument power consumption are exercised and power consumption calculated as in a) above. Instrument and probe power are turned off in preparation for the next test.

c) Instrument Turn-On. The test configuration is maintained and instrument power is turned on. The in-rush is photographed and from the photographs in-rush current is verified within specification limits. Instrument and probe power are turned off in preparation for the next test.

6) In-Rush Current

a) Probe Power Turn-On. Since the primary bus power is supplied to the instrument when probe power is on, it is desirable to check the in-rush surge current when probe power is applied to the instrument inertia circuitry. Excessive in-rush currents can degrade the solid-state fuses used on the probe to fuse the instrument power bus. The 0.1-ohm resistor is replaced with 1.0 ohm to increase measurement sensitivity and accuracy. Oscilloscope leads are placed across the resistor using 1X probes. Probe power is turned on and the in-rush current photographed. From the photograph, probe in-rush current is measured. The photograph becomes part of the test data package. Probe power is left on in preparation for the next test.

7) Interface Signals. The tests are designed to verify both instrument and probe interface signal parameters with both connected (under load). The series fuse box is removed and an interface test connector, cable and breakout box installed to provide access to each individual signal line. Probe power is turned on and the signal lines are checked as follows:

a) Power On/Off Line. The steady state value of this signal is checked with instrument power off using a digital volt meter. Instrument power is turned on and the on state value is recorded.

b) Analog Signals. All analog signals are then checked to verify operation. Values obtained under ambient test conditions are recorded.

c) Pulse or Digital Signals. Using an oscilloscope with 10X probes all pulse signals are measured to verify proper limits of rise time, fall time, amplitude, and duration.

d) Pulse Commands. Using an oscilloscope, proper command characteristics are verified for pulse commands regarding rise and fall time, amplitude, and duration.

8) Noise Tests. Probe or instrument generated noise transmitted on or coupled through to interface lines is measured for each instrument in turn with all other instruments disconnected. This test is performed in this way to provide a noise baseline since it has been found that as each instrument is connected, noise characteristics will vary. Using an oscilloscope, the

peak-to-peak noise on each interface line is recorded. Where specification limits are exceeded a photograph is taken and the amplitude and repetition rate of the noise recorded. A final noise measurement test is performed in the test described in the next section.

9) Instrument Data Check. Instrument and probe power is turned off. The interface test connectors and breakout box is removed and the probe harness mated to the instrument. Probe and instrument power are turned on. The probe and instrument are configured per specification requirements and a data check performed to verify proper operation of the instrument.

TASK 4: MAIN SCIENCE EQUIPMENT ASSEMBLY FUNCTIONAL TESTS

These tests, designed to verify instrument operation under most probe and instrument operating modes are detailed below:

- Probe turn-on and configuration to nominal flight configuration.
- Instrument turn-on, data check and calibration with and without stimulus as required.
- Instrument to instrument interference test wherein the instrument under test is configured to a preferred mode and all other instruments are exercised through power off/on and all operational modes to determine any interference between instruments.
- Probe to instrument interference wherein again the instrument under test is placed in a preferred mode. Probe communications and power equipment assembly subsystems are then operated throughout various modes as in flight to determine any interference with the instruments.
- Special instrument test requirements is a section of the test designed to provide additional testing unique for particular instruments such as special calibration of detectors, of bit rate, format tests.

TASK 5: MAIN SCIENCE EQUIPMENT ASSEMBLY SHELF TEST

The shelf-mounted science instruments will undergo random vibration and temperature cycling tests with the instrument electronics in an operating condition. Science instrument sensors, inlets, and appendages that mount directly to or through the pressure vessel will be attached for a nonfunctioning leak test of the probe and its interfaces at 25 atmospheres inward and then a combined pressure and temperature test.

TASK 6: INTEGRATION OF THE PROBE

Upon completion of the preceding tests the main science equipment assembly and the communications and power equipment assembly are physically integrated into the probe structure for integrated systems testing.

TASK 7: INTEGRATED SYSTEMS TEST NO. 1

The integrated system test is a combined probe/instrument test and is used throughout the test program to provide quick look reference data points to establish any instrument degradation throughout the program as well as subsequent to major systems test. The test itself is divided into two sections, engineering and science.

- The engineering test simulates a mission profile of launch, orientation and flight and verifies preparation of the probe subsystems.
- The science test simulates a flight profile. Instruments are exercised and stimulated with GSE, remote or onboard instrument stimuli to simulate flight data. It is recommended that the use of hardline GSE stimulus be minimized so as to maintain a simulated flight and simplify test operations.

TASK 8: INSTRUMENT ALIGNMENTS

Those instruments having optical alignment requirements are aligned to specification requirements using alignment fixtures designed and developed for this purpose. Alignments are performed at this time to verify integrity during the subsequent environmental test.

TASK 9: DESCENT PROFILE SIMULATION

The entire probe is installed in the hyperthermobaric chamber. Temperature and pressure are varied over the anticipated Venus atmosphere descent profile to surface conditions of 95 atmospheres and 927°F. All the science instruments will be operated in a mode simulating descent. Sensor stimuli and targets mounted on the chamber walls or on stub booms from the probe will be used in demonstrating the performance of the instruments integrated into the probe.

TASK 10: PROBE MAGNETICS TESTING*

The magnetic tests are a series of tests designed to determine the magnetic field of the probe at the position of the flight magnetometer sensor. These tests consist of the following:

- Probe functional test performed to verify probe operation subsequent to shipment to the TRW operated magnetic test site
- Probe operating modes
- Probe magnetization
- Probe demagnetization.

TASK 11: PROBE BUS AND PROBES INTEGRATION TESTING AT TRW

An extensive series of functional and environmental tests will be conducted to verify overall system compatibility, mass properties, EMC, probe checkout and deployment, solar vacuum, operations support, and software and DSN interface.

TASK 12: POST-ENVIRONMENTAL AND ETR INSTRUMENT TESTS

These tests are performed to provide instrument baseline operational data for comparison subsequent to shipment of the probes to ETR. They are similar to the detailed instrument tests described in Task 4 with the deletion of the interference portions of those tests.

TASK 13: INTEGRATED SYSTEMS TEST

An IST is performed at this time to verify overall systems operation and provide baseline data for the IST to be performed at ETR (see Task 7).

TASK 14: FINAL ALIGNMENT OF SCIENTIFIC INSTRUMENTS

If any of the scientific instruments are removed, realignment will be required.

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*For Version III science payload only. There are no magnetic requirements associated with the Version IV science payload.

ADDENDUM

OPTION TO SUPPLY PERSONNEL FOR INSTRUMENT DEVELOPMENT UNDER ARC DIRECTION

This addendum presents an option for additional tasks to be performed by TRW and MMC experiment engineers. These consist of instrument interface and integration related tasks, usually performed by ARC, for which an extensive knowledge of the spacecraft and probes at the system and subsystem level is desirable.

Implementation of the option would be by separate contract with ARC to clearly delineate the scope of the tasks, the time scheduling, and the technical direction of the personnel by the ARC Pioneer Venus experiment manager.

The optional tasks fall under the major categories shown in the dotted boxes in Figure A-1. Specific tasks which might be included under the option are:

- Experiment Design and Development
  - Monitor technical development of instrument
  - Generate and update instrument procurement specifications
  - Set up and maintain approved parts lists for instruments
- Experiment Test Support
  - Monitor development testing
  - Determine requirements for bench checkout equipment
  - Review instrument vendor test specifications, procedures, and plans
- Environmental Test Support
  - Monitor qualification and acceptance testing of instruments
  - Determine requirements for and insure compatibility of software for system testing
- Flight Operations Support
  - Determine requirements for mission operation software
  - Checkout mission operation software
  - Participate in mission operations in an instrument performance evaluation role.

An overall schematic diagram of these tasks, organized as a work flow chart to be compared with Figure A-2, is shown in Figure A-7, with the tasks again grouped into the categories of interface definition activities, documentation, and instrument integration and test activities. It is both possible and highly desirable to use one carefully selected set of contractor personnel to perform all these tasks, with continuity throughout all phases of the project. It is anticipated that the same personnel perform the spacecraft and probe-experiment related tasks. Early in the program, the option tasks are primarily concerned with instrument design, development, and testing; the concentration then shifts to assembly, calibration, integration, system testing, launch, and mission operations. Furthermore, the technical expertise required to perform all these tasks properly is the same — extensive knowledge of the instruments and the spacecraft/probes, the science objectives and ability to communicate effectively with representatives of the three key groups in the program: ARC, the experimenters, and the contractors.

The knowledge obtained by the contractor experiment engineers during the experiment design and development phase will serve to make them operate more effectively in performing their usual spacecraft/probe experiment tasks.

At present, it is believed that instrument development will have been underway for approximately 10 months before the instrument implementation phase begins; accordingly, one of the most critical sets of tasks is early and knowledgeable firming of the spacecraft- and probes-to-instrument interfaces. Contractor experiment engineers would be in the best position to firm up the interfaces (under the ARC experiment manager's technical direction) because of having worked through the system design study and having direct knowledge of, and access to, the workings of the project and the details of the spacecraft/probes system and subsystem characteristics. Such a procedure should significantly reduce overall costs by reducing the number of subsequent interface and design changes, which invariably have harmful cost and schedule effects, and by giving ARC tighter and more expeditious control over instrument design interfaces and more effective integration liaison. Although the rationale for this option is valid for orbiter, bus, and probes, it is

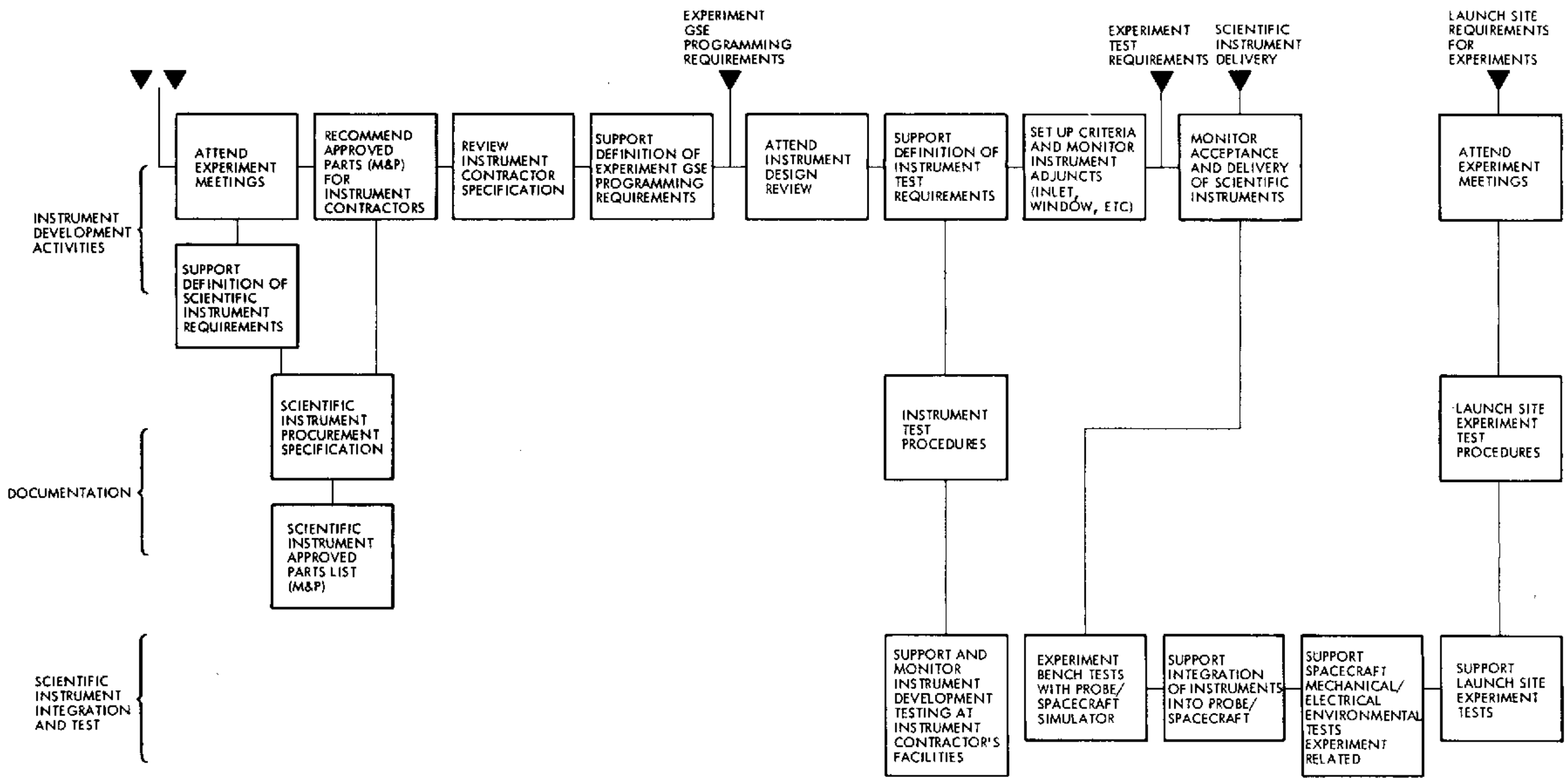

Figure A-7. Tasks Under Option to Supply Personnel for Scientific Instrument Development Under NASA/ARC Direction

particularly important for the probes. Many of the probe experiments and instruments utilize windows, inlets, ports, or structrual protrusions. These are neither all-instrument nor all-probe structure, but belong to both. Efficient development of the probes and instruments will be influenced considerably by the efficient firming of the requirements for and development of these adjuncts. It appears important to have a group of individuals who are both probe- and experiment-oriented specialists to work the optimum interfaces of these critical items.